DATA ACQUISITION SYSTEM FOR AIRCRAFT FLYING QUALITY INVESTIGATION

Robert Frederick Johnson

NAVAL POSTGRADUATE SCHOOL
Monterey, California

THESIS

DATA ACQUISITION SYSTEM
FOR
AIRCRAFT FLYING QUALITY INVESTIGATION

by

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Data Acquisition System
for
Aircraft Flying Quality Investigation

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ABSTRACT

Various methods of providing a data acquisition system for the Department of Aeronautics Cessna 310H model aircraft were considered. The system was to be reliable, functional and reasonably simple to operate in conjunction with its use in the flying laboratory. A need for dynamic flight information was recognized as an integral component in evaluating flying qualities. A three-axis rate gyro system was designed to meet this requirement. Output requirements included data accuracy, data availability and an associated simplicity in data reduction. A photo-panel data gathering system was selected to fulfill these requirements. The recording device used was a 35mm single-lens reflex camera system. Installation and component check-out of the data acquisition system was achieved at the Naval Postgraduate School during the period September 1973 to June 1974.

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I. INTRODUCTION

There exists in the field of Naval Aviation a continuing need for further development of existing concepts and a need to develop new ideas for systems becoming more and more complex. This need requires versatile and knowledgeable naval officers to evaluate the flying qualities of both refined and newly developed systems in the Naval Air Systems Command.

The Department of Aeronautics at the Naval Postgraduate School provides the specialized training for a significant number of officers to fill this requirement as a part of its graduate education program. An integral part of this specialized training occurs in the flight evaluation techniques courses, AE 3321 and AE 3322. It is during these courses that students evaluate the flying qualities in the department flying laboratory. This laboratory currently consists of a leased Cessna 310H model aircraft. Due to the rapid accumulation and complexity of data that must be gathered during flight it is necessary to have a data acquisition system. This system must include instrumentation for flight evaluation not normally found on an operational aircraft.

Investigation, research and installation of a photo-panel data acquisition system was performed at the Naval Postgraduate School, Monterey, California during the period September 1973 through June 1974.

II. REQUIREMENTS

The primary requirement was to design, engineer and install a data acquisition system for the Department of Aeronautics Cessna 310H flying laboratory. The data acquisition system was required to provide sufficient information for investigating the flying qualities of the aircraft. The complete system and the individual components that comprise it were evaluated and are discussed in their respective sections of this report.

Important criteria considered prior to finalizing the data acquisition system design included:

1. Number of component inputs.
2. Adaptability.
3. System design complexity.
4. Availability of components.
5. Financial justification
6. Reliability.
7. Maintainability.
8. Dynamic response.
9. Intricacy of data reduction.
10. Quality of data output.

The primary objective for installing a data acquisition system was to provide the student with a means of recording the data necessary to evaluate the flying qualities of an aircraft in an experimental environment. The output from the system was required to be both meaningful and readable.

It was necessary to install additional input information to sufficiently describe and evaluate the flying qualities. An integral part of the additional information required was accomplished by designing, evaluating and installing a rate gyro package as a data input. This package provides the user with roll, pitch and yaw rate information. Other inputs included control position and force measurement as well as airspeed, vertical speed, altitude, angle of attack, sideslip, temperature and elapsed time.

The system needed to be simple enough for student operation in that the flying laboratory is designed to evaluate flying performance and not the ability to operate a complex piece of machinery. The student needed simplicity of data reduction in order that more time be made available for actual performance investigation.

The installation of the system was intended to relieve the pilot and co-pilot from all responsibilities involved with data gathering. This enabled the pilot to fully concentrate on performing the required maneuvers which result in better data being made available for consideration. Another result is the increased safety factor involved with relieving the pilot of any recording function.

The system was required to be installed in a Cessna 310H aircraft which necessitated certain size, weight and electrical power restrictions. The system also had to achieve a combined satisfactory level of cost, maintainability and reliability.

In the interest of possible data acquisition expansion at some later date, the system was to provide the capability of adding additional inputs as deemed necessary.

III. INSTRUMENTATION

A. PHOTO-PANEL

Several data acquisition methods were considered before deciding to utilize a photo-panel. Relative cost, lack of extensive operational maintenance, system reliability and the simplicity of data reduction were factors considered before opting for the photo-panel method of data gathering. Previous use of photo-panel systems through the years employed movie type cameras with variable-burst film type recording. It was decided that a single-lens reflex type camera be used to attempt to obtain a higher quality of output resolution and to provide the user with a much simpler method of identifying the output events recorded. Burst type movie photography requires sorting a number of events from many feet of developed output whereas the single-lens reflex output contains a single identifiable frame of film per event.

The design of the photo-panel was dependent upon availability of components and size limitations involved with installation aboard the Department of Aeronautics Cessna 310H model aircraft. It was decided that the photo-panel theater would contain seventeen instruments.

1. Aileron Deflection

2. Elevator Deflection
3. Rudder Deflection
4. Aileron Force
5. Elevator Force
6. Rudder Force
7. Airspeed
8. Vertical Speed
9. Angle of Attack
10. Sideslip Angle
11. Temperature
12. Altimeter
13. Digital Voltmeter
14. Digital Voltmeter
15. Digital Voltmeter
16. Elapsed Time Clock
17. Event Counter

These instruments are mounted on a rectangular sheet of 0.125-inch thick 7075 T6 aluminum measuring 17.75 incles by 18.5 inches; the entire area being the minimum possible utilizing available instrumentation. The theater comprises the base for the photo-panel. The photo-panel theater is shown in Figure 2.

The photo-panel is rigidly mounted to the seat rails designed to support the port side passenger seat in the Cessna 310H. Four mounting brackets manufactured from 2024 T4 aluminum support the photo-panel through the theater face. The installed photo-panel system is shown in Figure 3.

The upper section of the photo-panel is a rectangular prism shape with a vertical height of 22.0 inches. The top plate was constructed of 0.160-inch thick 7075 T6 aluminum and supports the camera and camera control devices. The top plate and camera control system are illustrated in Figure 4. This section measures 12.5 inches by 12.0 inches. The photo-panel design was tapered as described to improve the reflected light options to be discussed later. The

height was a predetermined length which will also be discussed later.

The supporting frame for the photo-panel was constructed of 2024 T3 ninety-degree angled aluminum. The four side panels were constructed of 2024 T4 aluminum. One of these side panels is readily removable for maintenance purposes and also has a hinged door to allow access to instruments which must be preset prior to flight. The photo-panel system with removable side off is shown in Figure 5.

A 35mm Honeywell Pentax Spotmatic single-lens reflex camera was made available through the Naval Postgraduate School Photo Division of the Educational Media Department. In order to reduce the focal length required, and similarly the overall height of the photo-panel, a Super-Takumar 28mm f3.5 wide angle lens was obtained. This lens has a sixty-three-degree field of view and was the determining factor in selecting the minimum usable photo-panel height.

The photo-panel as described was completely enclosed to enable the camera to be used at a single constant exposure setting. The necessity of using light-emitting diode type digital voltmeters, see section B, caused a rather complex lighting problem. It was planned to use an available Honeywell Stronobar 600 photo flash unit with a very slow film. This was considered advantageous in that the flash unit operated on rechargeable batteries and would not further load the aircraft electrical system. This method produced fine reproductions of all instruments except the extremely

versatile and desirable digital voltmeters. The light-emitting diodes produced such a minute light intensity that it was impossible to filter the light or stop down the lens sufficiently to photographically record any output from the digital voltmeters. One other drawback which finally resulted in abandoning the flash unit concept was the thirty-second recovery time the unit would require between events.

It was determined that the only possible means of photographically recording information from these digital displays was by utilizing a red-sensitive high speed film with low intensity light within the photo-panel. It was determined that a relatively small light bulb would have to be utilized and the 24 VDC General Electric 1388 bulb was chosen. Two such lights produce 20 watts each at 1.75 amperes.

Two light-bracket assemblies were mounted on opposite sides of the photo-panel, designed so as to allow three degrees of freedom for adjusting each bulb. The bracket assembly is pictured in Figure 6. The interior walls of the photo-panel were painted flat white to allow non-glare light reflecting. The theater was painted flat black to absorb the reflected light. Each light assembly was shielded to prevent direct light glare and shadows on the instrument faces. A single rheostat was connected to the parallel light system and various light intensities, light reflecting angles, aperture settings and shutter speeds were tried before a satisfactory result was obtained. The test

apparatus utilized is shown in Figure 7. The final result eliminated the rheostat from the system and required a shutter speed of 1/125th of a second at an f-stop of 4.0. A sample output frame is provided in Figure 8. This result achieved several goals: The shutter speed was sufficiently fast to offset vibration effects, reasonably acceptable depth of field was available and the cumbersome rheostat was eliminated from the system.

The film chosen for the indicated operating conditions was Eastman Kodak Company High Speed Recording Film 2485 which is red sensitive and extremely fast. A standard light meter reading of eleven across the theater provides proper exposure at the chosen camera settings, light illumination and position. This film comes in bulk packages and can be obtained from the NPS Photo Division for various numbers of exposures. The exposed film is processed by the Photo Division and may be observed under an enlarging apparatus, or prints may be obtained for extracting data. This film must be developed in D19 for 13.0 minutes at 70.5 degrees Fahrenheit.

Prohibitive costs associated with obtaining an automatic film magazine for the camera necessitated designing a control system for advancing and actuating the camera with associated safety factors. The remote photo-panel control panel is mounted to the back of the co-pilot seat and operated by the individual occupying the starboard passenger seat in the Cessna 310H. The control panel is illustrated in Figure 9.

The operating sequence is commenced by actuating the film advance button. This signals a 27 VDC, 24 RPM, 0.5 ampere reversible motor designed for the aircraft 28 VDC electrical system. The motor rotates the required 160 degrees to advance the film and then returns to its starting position, actuating the ready light on the control panel and disabling the film advance circuit. Disabling precludes inadvertent camera damage. As soon as the ready light illuminates, the shutter actuation button may be depressed to record an event. The resulting signal momentarily freezes the digital displays through the inhibitor circuit and then actuates a 24 VDC intermittent duty push type Guardian Electric TP8X16 solenoid which actuates the shutter button on the camera. Upon release of the shutter release button the event counter is actuated and the film advance circuit is re-energized. The system is now ready for the next event to be recorded. Approximately four seconds are required from event to event, due mainly to the film advance motor speed.

The event counter in the theater is duplicated on the control panel for the operator to effectively monitor film usage and event sequencing and recording. The event counters are Minarik Electric Company series 7447 two-digit predetermining counters. They operate on 24 VDC at 0.125 ampere. The counters are set prior to flight, specifying the number of exposures available and automatically subtracting one digit each time an event is recorded. The control panel

operator need only specify maneuver type and event number to correlate the output.

Aircraft power at 28 VDC is supplied to the control panel and distributed by the photo-panel power switch, the inverter power switch, the regulator power switch and the digital panel-meter power switch. The photo-panel power switch illuminates the theater and provides power for the camera control system. The inverter power switch and the regulator power switch supply power to the rate gyro package as discussed in Section C. The digital panel-meter power switch provides power to the digital displays via a voltage regulator and heat sink apparatus to be discussed in Section B. All power switches are provided to secure the acquisition system in case of aircraft electrical malfunction.

The control panel also has three ten-position selector switches for use in conjunction with the digital voltmeters. Only two positions are presently utilized on each selector to provide roll, pitch and yaw rate information to the digital displays. This system is discussed in detail in Section C. The remaining eight positions on each selector may be used for future inputs to the system.

The camera is readily removed from the photo-panel package for protective storage or for film loading and unloading.

B. SENSORY RESPONSE INSTRUMENTATION

The sensory response instrumentation design phase required an investigation of what was necessary for analysis of static and dynamic aircraft parameters. From a previous photo-panel project and study, References 6 and 2 respectively, using the "Naval Air Test Center Fixed Wing Stability and Control Manual" and "NATC Performance Manual," the recommended sensory inputs were the control force and deflection inputs, angle of attack, sideslip, airspeed, altimeter and clock. Although these inputs were necessary, they were regarded as being incomplete for any thorough exploration of the dynamic flight regime. It was decided to add a rate gyro package, sensing pitch, roll and yaw rates, and a vertical speed indicator to the aircraft's sensory package. This provided the capability of examining to a greater extent the dynamic parameters.

Responses from the selected sensors had to be presented on instruments easily understood by any one collecting the data. Additionally, the output meters had to be either on hand or readily available. The result was a mixture of surplus indicators and meters from previously completed research work. The quality and operability of these instruments was resolved before further progress could be made. The accuracy of the photo-panel gauges was a question that could only be answered after a thorough investigation of each meter and its interface with existing sensory pickups had been performed.

1. <u>Electrical Meters</u>

Six meters for the control forces and positions were used with angle of attack and sideslip meters. These came from a flight instrumentation project that was terminated in the Spring of 1971. All were DC ammeters operating as galvanometers using a D'Arsonval movement connected to a pointer. The status and capabilities of these gauges were in doubt. In determining the internal impedance and full scale amperage rating, one meter was found to be inoperative. A replacement was obtained. Listed in Table I on the following page are the results of the examination.

Utilization of these meters exhibited advantages and disadvantages. The instruments presented response information in a form quickly recognizable to the student. All meter faces indicated explicitly the recorded output either as a specified control deflection in degrees left/right or trailing edge up/down or as a particular control force in pounds. Ease of altering the faces to meet the Cessna 310H output requirements facilitated adapting the instruments to their respective inputs. The inherent disadvantages were derived from the low electrical gauge resistance and/or amperage characteristics. The solutions to these problems will be discussed later.

Output range of the aircraft aileron deflection and elevator force necessitated redesigning the faces of the instruments. Black faces with white markings were used, identical with the remaining meters, thus producing

TABLE I

ELECTRICAL METER DEFLECTION AMPERAGE

AND INTERNAL IMPEDANCE

Meter	Amperage for full scale deflection in mA (positive & negative)	Internal impedance in ohms
Aileron Position	1.0	29.83
Elevator Position	0.5	21.70
Rudder Position	0.5	21.70
Aileron Force	1.0	150.80
Elevator Force	1.0	150.80
Rudder Force	1.0	160.60
Angle of Attack	0.025	1060.00
Sideslip Angle	0.052	1230.80

TABLE II

ELECTRICAL METER

RESPONSE SPECTRUM

Meter	Response position-degrees force-pounds
Aileron Position	+20/-20
Elevator Position	+25/-15
Rudder Position	+25/-25
Aileron Force	40 left/40 right
Elevator Force	70 pull/40 push
Rudder Force	150 left/150 right

a very readable scale. The scales were determined by the meter full scale deflections on either side of the zero point. The linear output of the aircraft sensor transducer made the transfer of definitive output to the meter displacement scale a simple ratio problem. The designed face was then finished in india ink and converted to a black face with white markings using a reverse negative photographic process. The face was glued to the instrument metal face plate.

In accordance with Reference 2 the response spectrum of the indicators must be at least as indicated by Table II on the preceeding page.

2. Existing Electrical Sensory Instrumentation

Positions of the aileron, rudder and elevator are sensed by a linear displacement potentiometric transducer that converts a change in the position of a movable contact (wiper arm) on a resistance element to a change in voltage. This change in response voltage is an indication of the displacement. Through a calibration procedure a correlation is established between the displacement and voltage.

Space Age Control Incorporated of Palmdale, California manufactured the three linear displacement transducers used on the aircraft. Each transducer is composed of a 2000-ohm rotating arm potentiometer enclosed in an aluminum housing. The rotating shaft of the sliding contact is connected to a spring-tensioned two-and-one-half turn spool enclosed within a drum mounted on the side of the potentiometer.

A flexible wire cable is wound on the spool and extends through an opening in the drum. This particular model was designed so that as the cable is unwound from the spool to a maximum of six inches, the slide contact resistance varies from zero to two thousand ohms, the full resistance of the pot. The maximum movement of any control linkage from its neutral position is plus or minus 2.375 inches. This is well within a useful transducer range of plus or minus three inches.

3. Control Forces

Forces exerted on the yoke and rudder are sensed by strain gauges positioned in the form of Wheatstone bridges. Internal resistance of the strain gauges varies according to the force being applied. This generates an electrical signal corresponding to the new resistive characteristic of the Wheatstone bridge. The co-pilot's yoke has two 350-ohm Wheatstone bridges; one for aileron forces and the other for elevator forces. Each bridge is composed of four C-9-171 type strain gauges. Two strain gauge transducers manufactured by Radiation Incorporated of Los Angeles, California are attached to the rudder pedals, one for each pedal, for rudder force pickup.

4. Flight Boom Vane

Sideslip and angle of attack sensor inputs are received from a yaw and pitch sensor (YAPS) head that was obtained from the Army Flight Test Facility, Edwards Air Force Base, California. Two aluminum vanes are mounted

perpendicular to each other on the vane housing and are deflected by the relative wind. The vanes are connected to individual one-turn rotary potentiometers enclosed within the boom head. The rotary potentiometer acts as an angular displacement transducer with the vane deflection changing the resistance, causing an appropriate change in the output voltage.

5. Junction Box

Voltage distribution requirements of the individual sensors and the succeeding response collection is satisfied by the aircraft junction, "J," box. The aircraft electrical system generates 28 VDC. By a system of voltage regulators the 28 VDC is reduced down to either 5 VDC or 15 VDC excitation voltage as necessary for the five individual potentiometric transducers; three control deflections, angle of attack and sideslip. All output circuitry from the five transducers are channeled back to their lead terminals in the "J" box. Connected in a bridge with each sensor pickup is a balancing potentiometer, useful for calibrating and establishing zero reference points. The location of each adjustment balancing nut is indicated on the junction box. See Figure 17.

Wheatstone bridges in the three force sensors constituted different voltage and output collection specifications from that of the five potentiometric devices. The response signal transmitted by a bridge is quite low and must be amplified to be a useful indication. A Grant model

DCA8-3 operational amplifier is used for the necessary output signal magnification. The operational amplifier receives 28 VDC power from the junction box and, through a gain setting, relays a usable output signal spectrum of from a negative 0.695 VDC to a positive 5.0 VDC. The amplifier supplies a constant five-volt excitation signal across the Wheatstone bridge. Four Grant DCA8-3 operational amplifiers are used, one each for the aileron and elevator force sensors and two for the rudder force transducers.

6. Interfacing of Ammeters and Sensory Systems

From previous research completed on the aircraft sensory systems, a set of tables and graphs relating input measurements to output voltages were available. A linear output response was exhibited for each sensor except the sideslip, which had nonlinearities for all left deflection values.

An investigation was conducted on each meter to determine if its responses were linear and if the voltage range was satisfactory. All ammeters are linear devices but operate in the zero to 160 millivolt range. The extremely low instrument voltage range induced the utilization of either a shunt resistor or a trim potentiometer on the back of each indicator. This guaranteed that the sensor output voltage would not saturate the instruments. As the meters were linear, the mating of the indicator to its input produced a linear result.

7. Calibration

Individual assessment of each meter to its specific sensor system and input was performed. Calibration of the instruments was accomplished by dividing the gauges into three categories: control deflection group, angle of attack and sideslip, and force sensors.

Calibration of the control deflections was accomplished totally at the aircraft. Each meter was connected to its respective output lead at the junction box and a variable resistance decade box, simulating the shunt resistor, was placed in parallel with the meter. A specific deflection was set either by a clinometer for the aileron and elevator or a scribed metal plate for the rudder. By varying the decade box the correct linear response was achieved, matching the meter indication to the actual control position. Several complete output cycles were performed to determine the response repeatibility and hysteresis. The results are shown in Figures 17, 18 and 19.

Interfacing of the YAPS boom vane and the angle of attack and sideslip meters was accomplished in the laboratory as the boom was not yet attached to the aircraft. Simulation of the aircraft junction box and its balancing potentiometers was performed by using a regulated power supply, set at 15 VDC, and a 5000-ohm ten-turn potentiometer. The individual meter was placed between its corresponding rotary transducer and balancing potentiometer, which formed a bridge. A decade box was then introduced in parallel with

the meter, acting as a shunt resistor. A scribed aluminum plate located around the vane shaft permitted a range of specified angles for meter measurements. Repeatibility and hysteresis were investigated by completing several cycles after the initial calibration was made. See Figures 20 and 21 for calibration curves.

Force meter calibrations were partially simulated in the laboratory and completed at the aircraft. The laboratory portion of the investigation was vital for the adaption of the instruments to the operational amplifiers and Wheatstone bridge sensors. Grant DCA8-3 amplifiers are designed to magnify voltage in one direction only; the meters with a center-position null setting need dual polarity voltages. There existed a discontinuity between what the gauge required and what the amplifier produced. A four leg 350-ohm Wheatstone bridge was constructed and connected to a Grant DCA8-3 amplifier. Twenty-eight VDC was supplied to the amplifier and its output signal was passed on to a force indication meter. A 330K-ohm resistor was used to simulate strain readings across a leg of the bridge. In series with the meter was a trim potentiometer with the size determined by experimentation. By rotating the 330K-ohm resistor across opposite legs of the bridge, different signal polarity outputs were generated. Different size trim potentiometers were used until the voltage polarity and maximum needle deflection were within the limits of the gauge. This provided a trim potentiometer size which cancelled out the amplifier and meter discontinuity.

With compatibility requirements of meters and sensors satisfied, calibration of the system on the aircraft was approached. In past research, force sensing systems were assessed in the laboratory as this was the most accurate and easiest method. System installation was accomplished afterward. But with the sensing device previously positioned, a calibration technique was devised for accomplishing this at the aircraft. A desired benefit from this procedure is the addition of pre- and post-flight calibration checks on sensitive laboratory flights.

The calibration method had to incorporate the provision of having the strain gauges sense only an applied force. This was accomplished by locking the aircraft yokes and control surfaces and then applying a load to the force transducer under examination. Application of a force to only one strain bridge presented difficulties for the aileron and elevator measurements, as they are on the same yoke. For the aileron calibration a cable pully weight arrangement was constructed that applied the desired moment to the aileron strain bridge only. The push or pull load conditions were obtained on the elevator by using a metal adapting bar positioned across the horizontal center line of the yoke and attached at each end. Then a chatillon variable force meter was used to load the elevator control. Rudder pedal transducers were assessed by the application of measured weights directly to the strain bridge transducers. Results of calibration procedures, repeatibility and hysteresis cycles are shown in Figures 22, 23 and 24.

8. <u>Pressure Instruments</u>

Three pressure instruments were placed in the photopanel: airspeed indicator, altimeter, and a vertical speed indicator. The VSI was obtained from the Army Flight Test Facility, Edwards Air Force Base, California, and the remaining pressure instruments were from departmental supplies.

The airspeed indicator used contains sensitive pressure elements which indicate the difference between static and pitot pressures. The static and pitot pressures, transmitted by nylon tubing from the port wing pitot-static tube, are introduced to opposite sides of a diaphragm housed in the instrument. Slight movements of the diaphragm, caused by changes in either pitot or static pressures, are transmitted to the pointer which indicates corresponding airspeeds on a zero- to 400-knot scale. The airspeed indicator was calibrated in the Naval Postgraduate School subsonic wind tunnel.

The altimeter is a static pressure sensing instrument capable of indicating barometric altitude in tens, hundreds, and thousands of feet by using three pointers. Static pressure from the port wing pitot-static tube enters an evacuated diaphragm assembly and through a connecting deflection diaphragm transmits static pressure changes to the three pointers. Field barometric pressure readings are set and displayed in the Kollsman window. This information is used by a system of gears in the altimeter

to correct for differences arising from using a reference sea level pressure other than the standard 29.92 inches of mercury.

The vertical speed indicator is a static pressure sensing indicator. Static pressure from the port wing pitot static tube is fed to a diaphragm and the VSI case volume surrounding the diaphragm. The rate of climb or descent causes the static pressure to change differently within the diaphragm than within the case. This rate of pressure change sensed by the diaphragm is transmitted to a pointer. The pointer indicates in hundreds of feet per minute the rate of climb or descent. Accuracy of the VSI must be flight checked. The confidence attached to its reliability and quality is high since it was calibrated by the Army laboratory prior to its shipment.

The port wing pitot-static tube was not the primary selection as a pressure source for the three pressure instruments. The YAPS head, with its own pitot static tube positioned well ahead of the aircraft and outside any pressure influence patterns, was originally selected as the pressure source. Due to delays in adapting the boom to the aircraft, an alternate selection of the port wing pressure source was made. The use of this location as a static source for a light aircraft flying 200 knots or less will generate no appreciable errors in the static pressure readings, according to Reference 1. The pitot pressure accuracy in this location should be explored by comparing the pilot's airspeed indication

with that indicated in the photo-panel for a range of values and attitudes.

9. Digital Panel-Meter

It was decided to incorporate digital voltmeters in the photo-panel theater for the recording of rate information. The photo-panel system required that these instruments be both small in size and highly reliable. Three such instruments were needed in order to simultaneously record information about the aircraft's three axes. This requirement was satisfied by obtaining three Datel Systems Model DM-1000 digital panel-meters from a previous research project. An enlarged digital panel-meter is illustrated in Figure 10.

The DM-1000 is 3.0 inches wide, 1.75 inches high and 2.25 inches deep, thus satisfying the size requirement. They individually weigh only six ounces, and past performance had proven them to be highly reliable. The digital display readout ranges from negative to positive 1.999 volts. Each panel-meter requires 5.0 VDC at 0.600 amperes of aircraft power. Since the aircraft has a 28 VDC electrical system it is necessary to dissipate a large power loss in the voltage regulator due to the large current the digital panel-meter requires to properly function. Heat sinks are provided for each voltage regulator and are mounted in an aluminum box on one side of the photo-panel. See Figure 11.

The three ten-position selector switches mounted on the photo-panel control panel determine what information is linked to the digital panel-meters. Refer to Section C for present usage. Due to extremely high sampling rate inherent to these meters, an inhibitor circuit is utilized when data gathering. This inhibitor circuit is an integral part of the camera control circuitry and functions automatically.

As referenced in Section A, a critical problem was encountered by utilizing this particular meter. This problem was caused by the use of the light emitting diode or LED display. This is a galium arsenide phosphide display that is, although readily readable by the human eye, of a very low light intensity. The low light intensity emission posed a critical problem for photographic recording. This problem was resolved through experimentation as indicated in Section A. Although the light intensity is marginal, the red circular polarizers provided with each unit display a clear readout.

C. RATE GYROS

The miniature rate gyro, Honeywell type GG16C, is a small hermetically sealed, gyroscopic instrument measuring angular velocity. It is filled with a viscous fluid to provide a damping medium and partial flotation of the gimbal. Distinguishing features include a 2400 RPM, three-phase 400-hertz spin motor. Two potentiometer turn-rate measurement pickoffs are fixed on the gimbal axis. Fast

warmup to operating temperature is provided by an integral heater controlled by a thermostatic switch. Maximum input turn-rate for this type rate gyro is 55 degrees per second.

The power requirements are as follows:

1. Spin motor: three-phase, 400-hertz plus or minus one per cent, 26 plus or minus 0.5 volts per phase (line to line) for start and run, and an A-B-C sequence line for phase rotation.

2. Heater power: single phase, 115 plus or minus one volt, 400-hertz, for initial warmup heater excitation power will be 150 watts.

3. Front differential potentiometer (Pot 1): five VDC with the maximum resistance across the potentiometer being 3000 ohms.

4. Rear potentiometer (Pot 2): two- to five-VDC with the maximum resistance across the potentiometer being 12750 ohms for serials L-165 and L-71 and 13250 ohms for serial L-24.

5. Potentiometric output signal: both potentiometers will have a response range fifty per cent of the excitation voltage value.

Initial investigation of the rate gyros commenced with an exploration of their present condition. A 400-hertz, three-phase AC inverter with a rated output of 500 VA at 115 VAC was obtained as a power supply. Two variacs were used for reducing the 115-VAC output of two phases down to 26 VAC for the spin motor. A five-VDC power supply was introduced across the rear potentiometer and its output channeled to a digital voltmeter. Condition of the gyroscopic mechanism was determined by giving the instrument a random angular velocity input and then steadying the case. The results verified that the rate gyros are functional and their inner sensing pickoffs were performing.

Preliminary steps were taken to calibrate the rate gyros and install them as a package in the aircraft. The necessary retainer for locking the cases in an orthogonal axis reference frame was designed. An adequate AC inverter, compatible to the aircraft in both size and power requirements, and to the rate gyro package electrical specifications, was required. The present AC inverter was too large and necessitated an input power of 28 VDC and 38 amperes for its operation. The aircraft electrical system produced 28 VDC but could only distribute ten amperes. In addition to a new AC inverter, three 115 VAC and 26 VAC transformers were needed that were small and readily adaptable to the aircraft.

1. Calibration

The procedures for an accurate calibration of the package were examined. A mechanism was needed that could turn at very low speeds. This low turning rate was assumed, based on the fact that a standard rate turn is three degrees per second or one-half revolution per minute. Validation runs concentrated on angular velocities from three degrees per second to eighteen degrees per second or from one-half to three revolutions per minute. A turn table powered by an electric shaft motor through a varying series of gears was the means selected to create the angular rates. Such a rate table was designed and the Aeronautics Department metal shop manufactured the table and accessory gears. An electric drive shaft motor with a very low RPM shaft output, 19 RPM, was used to drive the table.

All three gyros must receive spin motor and heater power at the same time. For test purposes a brass board model of what was to appear in the aircraft was constructed. A DC-regulated power supply was used for the potentiometric input voltage. The potentiometer signal output was wired via a wire junction plate to a strip chart recorder and a digital voltmeter. The rate gyro package orthogonal retainer was then placed on the turn table. All leads to and from the gyro package were collected on the wire junction plate held suspended over the calibration table. This permitted relatively good turning freedom for the input table. See Figure 43 for the calibration setup.

Before actuation of the table, a final check was made to ensure that the input axis around which the gyroscopic mechanism sensed its angular rate was positioned correctly. The input axis is indicated on the gyro case lip as a notch. This permits the location of the input axis to be 90.0° plus or minus 0.2° from the plane of rotation, as called for in the specifications for proper rate measurements.

The correct gyro for sensing the turntable input was then calibrated. Angular velocities of 3.00, 4.76, 6.24, 8.90, 12.00, and 17.89 degrees per second were used as the specified values. After all three gyro outputs were recorded, an examination of the orthogonal quality of the package was performed. This demonstrated that the rate gyros were at right angles to each other, sensing angular rates about their

individual axes and none other. A summary of results for the calibration runs are found in Figures 25 through 34.

Several facts were needed before interpreting the strip chart recording measurements as shown in Figures 30 and 31. Positive rotation for the rate table was defined as the rotation dictated by the right hand rule about the corresponding body axis for positive angular velocity. The gearing system used on the assessment table caused the rotation to change direction for values of 6.24 through 17.89 degrees per second from those of 3.00 and 4.76 degrees per second.

For all measurement runs the first two values are for positive rotation and the next four represent a negative input. For linearity investigations and calibration values all outputs were treated as positive. This could have been duplicated on the strip chart by reversing the input voltage polarity. The null position for the differential potentiometers varied slightly, whereas each rear potentiometer has a unique null setting. The chart recording used the null value as its zero reference point vice an actual 0.000 voltage.

Listed in Table III on the following page are the results of the turntable angular velocity inputs. Note that all output readings are in volts. All outputs exhibited good linearity from 6.24 degrees per second onward. The slight variation in linearity noted by the potentiometers, Pot 1 and 2, at the 3.00 and 4.76 degree per second values is due to the threshold sensitivity of the gyros. The differential

TABLE III

TURN TABLE ANGULAR VELOCITY

OUTPUT CALIBRATION RESULTS

Rate gyro	Pot/Input and null settings (input VDC)		Inputs (degrees per second) 3.00	4.76	6.24	8.90	12.00	17.89
			Output signals (volts)					
Pitch	1/5.0	-.013	.100	.170	.260	.380	.520	.765
Pitch	2/5.0	+.650	.030	.045	.065	.090	.125	.215
Roll	1/5.0	+.042	.120	.200	.260	.380	.520	.780
Roll	2/		Inoperative					
Yaw	1/5.0	-.002	.085	.150	.240	.375	.520	.750
Yaw	2/2.0	+.998	.048	.065	.110	.155	.210	.320

potentiometer has a 0.99 degree per second minimum rate for an excitation voltage of 50 VDC and the rear potentiometer threshold rate at an input voltage of 100 VDC is 0.79 degrees per second. By using an appreciably lower excitation voltage as dictated by the aircraft electrical system and output limitations of the recording meters, positive to negative 1.999 VDC, slight accuracy is lost at the lower angular rates. This loss is approximately one degree per second.

A range of output voltages for input angular velocities were extrapolated from the calibration data. See Figures 28 and 29. The three differential potentiometers presented close correlation, except as noted previously for threshold sensitivity. The small differences at the 17.89 degree per second value are due to wire cable binding at a rapid turning rate, thus generating slight opposition of the retainer package to the input rotation.

2. Aircraft Adaption and Installation

The rate gyro package unit designed and built for the Cessna 310H is composed of four major subsystems: the three rate gyros and retainer and AC inverter, along with AC and DC distribution boxes. The four sybsystems are mounted on a 17- by 22-inch 2024 T6 aluminum plate fastened to the four tie-down positions of the aft-most right-hand seat area. The plate is elevated 6.7 degrees up, relative to the seat rail plane, for alignment of the rate gyros' retainer to the X-Y plane of the aircraft body axis. The retainer is positioned on the plate parallel to the aircraft

longitudinal axis. This was achieved by knowing the angle, 3.78 degrees, between the longitudinal axis and a line drawn through the outboard two tie-down holes. The package is correctly positioned according to the aircraft's body axes.

Heater and spin motor power is provided by a small three-phase, 400-hertz AC inverter measuring seven inches by three inches by five inches, with an output rating of 100 VA. The inverter output is directed to the AC distribution box. The AC box contains three 115-VAC to 26-VAC transformers arranged in a delta group for spin motor requirements and associated circuitry for the heaters' 115 VAC needs. Located on the face of the box are three lights indicating the status of heaters and spin motor phases. When light A is on, phase A 115 VAC heater power is being supplied to the three gyros. The illumination of all lights represents the application of A, B and C phases to the spin motors.

The varied voltage requirements of the sensing potentiometers necessitated a DC distribution box. The three differential pickoffs are supplied by individual 5-VDC voltage regulators from aircraft 28-VDC power via the photo-panel control panel. A similar system is used for the rear potentiometers with a modification adjusting the regulator output voltage. Reduction in excitation was necessary to keep the rear sensor response below 1.999 VDC, the recording meter upper limit. A 50K-ohm trim potentiometer, as the

modifier, is located in series with Pot 2 regulator output. The corresponding adjustment nut is indicated on the distribution box.

A pre-heat function of the rate gyro package is required. The heaters, in raising the viscous fluid up to an operating temperature, draw a high amperage load from the AC inverter of between eight to ten amperes. This is reaching the maximum amperage output for the junction box. A pre-heat capability is built into the system for pre-flight preparations. Once the viscous fluid operating temperature is reached, the heaters draw less than one ampere for their continuous running. Two items are used for the pre-heating of the package: an external 28-VDC ground power source and a two-position control switch marked "heater" on the AC chassis box. The switch's two positions are "off" and "115 VAC" heater power. A DC ammeter is located in series with the ground-support 28-VDC input line. When the amperage drops from eight-to-ten down to one ampere, the preheating is completed. Disconnecting the ground-located 28-VDC supply is accomplished, and reintroduction of the normal power cable is performed.

IV. CONCLUSIONS AND RECOMMENDATIONS

The data acquisition system which was installed in the Department of Aeronautics Cessna 310H is entirely capable of providing the information required for investigating the flying qualities of the aircraft. The incorporation of the rate gyro package tremendously increased the amount of dynamic information available. This particular additional input should provide the student, in the flight evaluation courses offered, with a far better understanding of aircraft stability and control. The rate gyro system is also a single entity and may be quickly adapted for use in any vehicle requiring the collection of pitch, roll and yaw rate information.

Preflight and normal operating procedures, as presented in Appendix C, are relatively simple as proposed. The insignificant increase in time required to ready the system for flight is more than compensated by the quantity, quality and ease of recording data. The completed system fulfilled the design goal of providing data from a system complex enough to provide accuracy yet simple enough to avoid costly and time consuming maintenance.

The sensory response instrumentation provides the student with accurate and easily recognizable aircraft parameters. The calibration and installation procedures described in this report ensure the quality of results. The out data are simple to reduce in order to allow the

student more time for analyzing results. Correlation of data output is easily accomplished through the use of the event counters installed in the photo-panel and the control panel.

The digital panel-meters utilized in the photo-panel should incorporate a stronger display-light output when one becomes available. A stronger display-light intensity would greatly enhance the quality of the film output. At present, this quality is good, but if light from the digital panel-meter displays were increased the developing process could be altered to reduce or eliminate the existing grain in the output.

Greater input angular velocity could be used by installing slip rings on the rate gyro calibration apparatus. This would eliminate retainer opposition to input rotation caused, at present, by direct wire bundle connections.

A single-phase 400-hertz 115-VAC inverter for pre-heat power to the AC chassis box is needed. This will prevent premature failure of the present rate gyro package inverter due to high load conditions during the pre-heat stage.

Another recommendation for improving the system is to replace the Grant DC8-3 operational amplifiers. These components have an inherent zero-point drift and are not bipolar.

It is recommended that the inclusion of a normal acceleration input be made. This addition could easily be accomplished through one of the control panel multiposition switches to a digital panel-meter. Only six of the thirty

total positions available on the control panel rotary switches are presently used, allowing for future system flexibility.

An outside air temperature probe is needed. A linear thermistor was purchased and is available for system use. An outside air temperature gauge is presently installed in the photo-panel but is not operational.

It is imperative that an individual be assigned responsibility for system upkeep; this would include the capability of ensuring that the system remains operationally ready. Maintenance was not intended to be, nor is it, an extremely time consuming factor. It is considered important, however, so as to gain the maximum utilization possible.

APPENDIX A

WIRING RUN SHEET

Rate Gyro Wiring Run

Pin connector No. 28-4501-32P   Plug end
Pin connector No. 28-4501-32S   Adapter end

Located on top of rate gyro package container.

Plug pin terminal	Wire color from gyros	Comment	Adapter pin terminal	Wire color to AC & DC boxes
1	blue	heater and motor grnd	1	black
2	red	115 VAC heater pwr		
3	red	115 VAC heater pwr	3	clear
4	red	115 VAC heater pwr		
5	white/ orange	roll pot 1 0.0 volts	5	white
6	green	roll pot 1 low out	6	green
7	white/ brown	roll pot 1 hi out	7	black
8	white/ red	roll pot 1 +5.0 volts	8	red
9	black	roll pot 2 low out	9	black
10	white	roll pot 2 +2.0 volts	10	white
11	white/ green	roll pot 2 hi out	11	green
12	white/ blue	roll pot 2 0.0 volts	12	red

13	white/ orange	pitch pot 1 +5.0 volts	13	white
14	green	pitch pot 1 low out	14	green
15	white/ brown	pitch pot 1 hi out	15	black
16	white/ red	pitch pot 1 0.0 volts	16	red
17	white/ violet	26 VAC phase B		
18	white/ violet	26 VAC phase B	18	clear
19	white/ violet	26 VAC phase B		
20	white/ yellow	26 VAC phase A		
21	white/ yellow	26 VAC phase A	21	red
22	white/ yellow	26 VAC phase A		
23	white	yaw pot 2 +2.0 volts	23	clear
24	white/ green	yaw pot 2 hi out	24	red
25	white/ blue	yaw pot 2 low in/out	25	black
26	white/ orange	yaw pot 1 0.0 volts	26	white
27	green	yaw pot 1 low out	27	green
28	white/ brown	yaw pot 1 hi out	28	black
29	white/ red	yaw pot 1 +5.0 volts	29	red
30	white	pitch pot 2 0.0 volts	30	clear

31	white/ green	pitch pot 2 hi out	31	red
32	white/ blue	pitch pot 2 low in/out	32	black

AC Distribution Chassis Wiring Run

Pin connector No. MS 3106A-16S-1P

Located on side of AC distribution chassis.
Used for AC power distribution to rate gyros.

Terminal letter	Wire color from gyro retainer	Comment
A	black	phase C motor grnd
B	clear	phase B 26 VAC
C	red	phase A 26 VAC
D	black	phase C heater grnd
E		spare
F	clear	phase A 115 VAC heater pwr

Pin connector No. MS 3106A-14S-5P

Located on side of AC chassis
Used for AC inverter power input to AC chassis box.

Terminal letter	Wire color from AC inverter	Comment
A	red	phase A 115 VAC
B	green	phase C 115 VAC
C		spare
D	black	phase C grnd
E		spare

Note: AC inverter phase output is A-C-B. Rate gyro phase input requirement is an A-B-C sequence. Therefore, a renaming of the inverter phases is done for clarification. Inverter output phase C is called phase B and inverter output phase B is called phase C.

DC Distribution Chassis Wiring Run

1. Input/output cable from roll rate gyro.

Pin connector No. MS 3106A18-1P

Located on side of DC distribution chassis and indicated by a roll label.

Terminal letter	Wire color from gyro retainer	Comment	Internal wire color
A		shield	
B	white	pot 2 +2.0 volts	orange
C	black	pot 2 low out	purple
D	red	pot 2 0.0 volts	black
E	white	pot 1 0.0 volts	black
F	red	pot 1 +5.0 volts	yellow
G	green	pot 1 low out	gray
H	black	pot 1 hi out	purple
I		spare	
J	green	pot 2 hi out	white

2. Input/output cable from pitch rate gyro.

Pin connector No. MS 3106A18-1P

Located on side of DC distribution chassis and indicated by a pitch label.

Terminal letter	Wire color from gyro retainer	Comment		Internal wire color
A		shield		
B	clear	pot 2	+5.0 volts	orange
C	red	pot 2	hi out	blue
D	black	pot 2	0.0 volts	black
E	red	pot 1	0.0 volts	black
F	white	pot 1	+5.0 volts	yellow
G	green	pot 1	low out	gray
H	black	pot 1	hi out	blue
I		spare		
J		spare		

3. Input/output cable from yaw rate gyro.

Pin connector No. MS 3106A18-1SIC

Located on side of DC distribution chassis and indicated by a yaw label.

Terminal letter	Wire color from gyro retainer	Comment		Internal wire color
A	black	pot 1	hi out	green
B	green	pot 1	low out	gray
C	red	pot 1	+5.0 volts	yellow
D	white	pot 1	0.0 volts	black
E	black	pot 2	low out	black
F	red	pot 2	hi out	green
G	clear	pot 2	+2.0 volts	orange
H		shield		

Photo-Panel Theater Wiring Run

Pin connector No. 57-80500

Located on side at base of photo-panel

Terminal number	Internal wire color	Wire leads to
1	black	Pin 18A DPM # 3
2	green	Pin 1A DPM # 3
3	clear	Pin 2B DPM # 3
4	black	Pin 18A DPM # 2
5	green	Pin 1A DPM # 2
6	clear	Pin 2B DPM # 2
7		spare
8	black	Pin 18A DPM # 1
9	green	Pin 1A DPM # 1
10	clear	Pin 2B DPM # 1
11		spare
12		spare
13		spare
14		spare
15		spare
16		spare
17		spare
18	black	aileron position right
19	black	elevator position right
20	black	rudder position right
21	black	aileron force right
22	black	elevator force right
23	black	rudder force right

24	black	sideslip
25	black	angle of attack
26	red	Pin 18B DPM # 3
27	white	Pin 1B DPM # 3
28	orange	Pin 10A DPM # 3
29	red	Pin 18B DPM # 2
30	white	Pin 1B DPM # 2
31	orange	Pin 10A DPM # 2
32		spare
33	red	Pin 18B DPM # 1
34	white	Pin 1B DPM # 1
35	orange	Pin 10A DPM # 1
36		spare
37		spare
38		spare
39		spare
40		spare
41		spare
42		spare
43	red	aileron position left
44	red	elevator position left
45	red	rudder position left
46	red	aileron force left
47	red	elevator force left
48	red	rudder force left
49	red	sideslip
50	red	angle of attack

Camera Control Wiring Run

Pin connector No. MS 1306A18-1S

Located on top of photo-panel

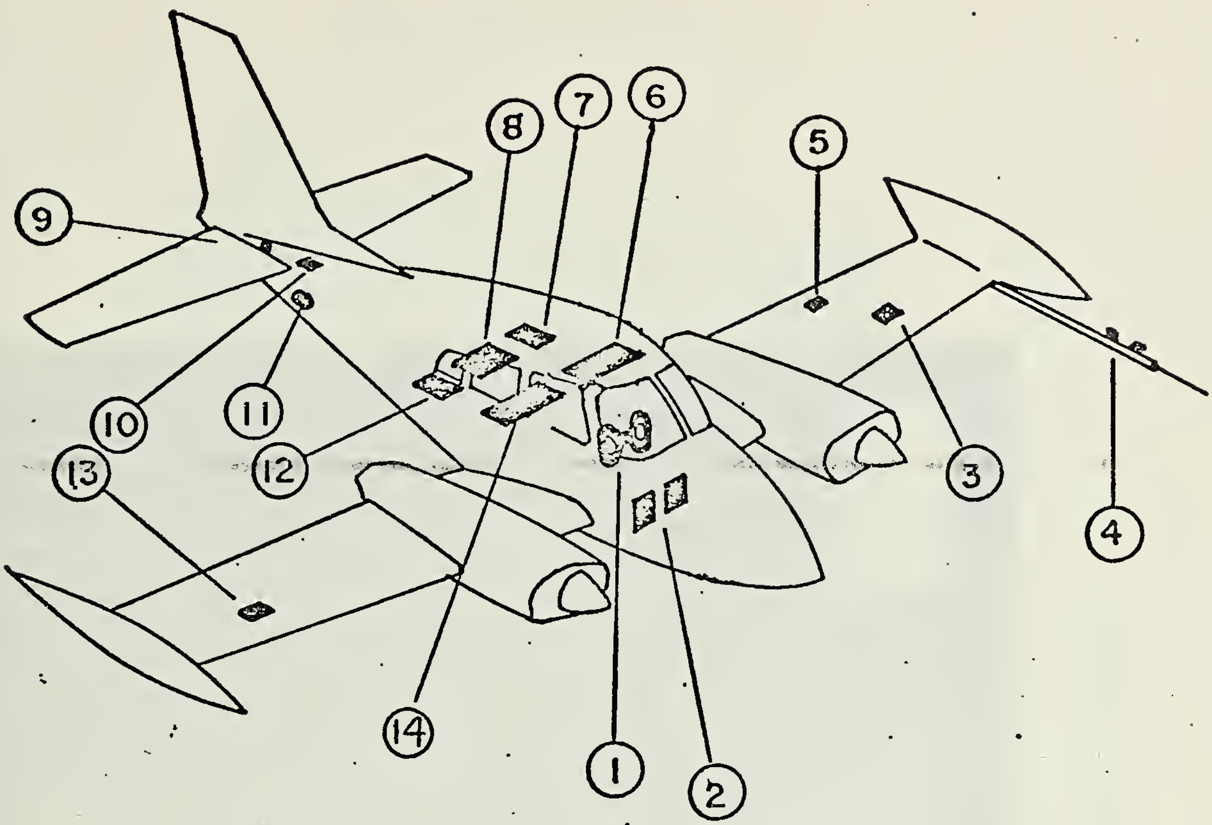
Terminal letter	Internal wire color	Wire leads to	Comment
A	red	TB-10	+28 VDC in
B	black	TB-11, 12	-28 VDC in
C	purple	TB-4	
D	white	TB-7	
E	orange	TB-10	
F	black	solenoid	
G	orange	P7/RL3	
H	black	TB-11, 12	
I	white	stop MS	
J			spare

Terminal Board

Located on top of photo-panel

Left Side			Right Side	
Wires lead to	Wire color	Terminal number	Wire color	Wires lead to
P9/RL1	red	1	purple	lorev MS
P7/RL1&2	purple	2	blue/ red	lorev MS/ hirev MS
P6/RL1	red	3	red	hirev MS
P9/RL2&3	red/ red	4	purple	CP C
P6/RL2	red	5	black	stop MS
P6/RL3	red	6	white	light off MS

TB-10	purple	7	white	CP D & light off MS
P1&4/RL1	brown	8	black	film advance motor
P5&8/RL1	gray	9	yellow	film advance motor
CP A	red	10	purple/ purple/ purple	P1/RL3 light off MS ready light
CP B	black	11	black/ black black	P3&10&2/RL1 P3&10/RL2 P3&10/RL3
CP H	white	12	TB-11	



1. WHEEL FORCES
2. RUDDER FORCE
3. PORT PITOT TUBE
4. FLIGHT BOOM
5. AILERON POSITION
6. PHOTO-PANEL
7. ACCELEROMETER
8. J-BOX
9. ELEVATOR POSITION
10. RUDDER POSITION
11. TRAILING CONE
12. RATE GYRO PACKAGE
13. STBD. PITOT TUBE
14. PHOTO-PANEL CONTROL BOX

FIGURE 1
CESSNA 310H INSTRUMENTATION COMPONENTS

APPENDIX B
FIGURES

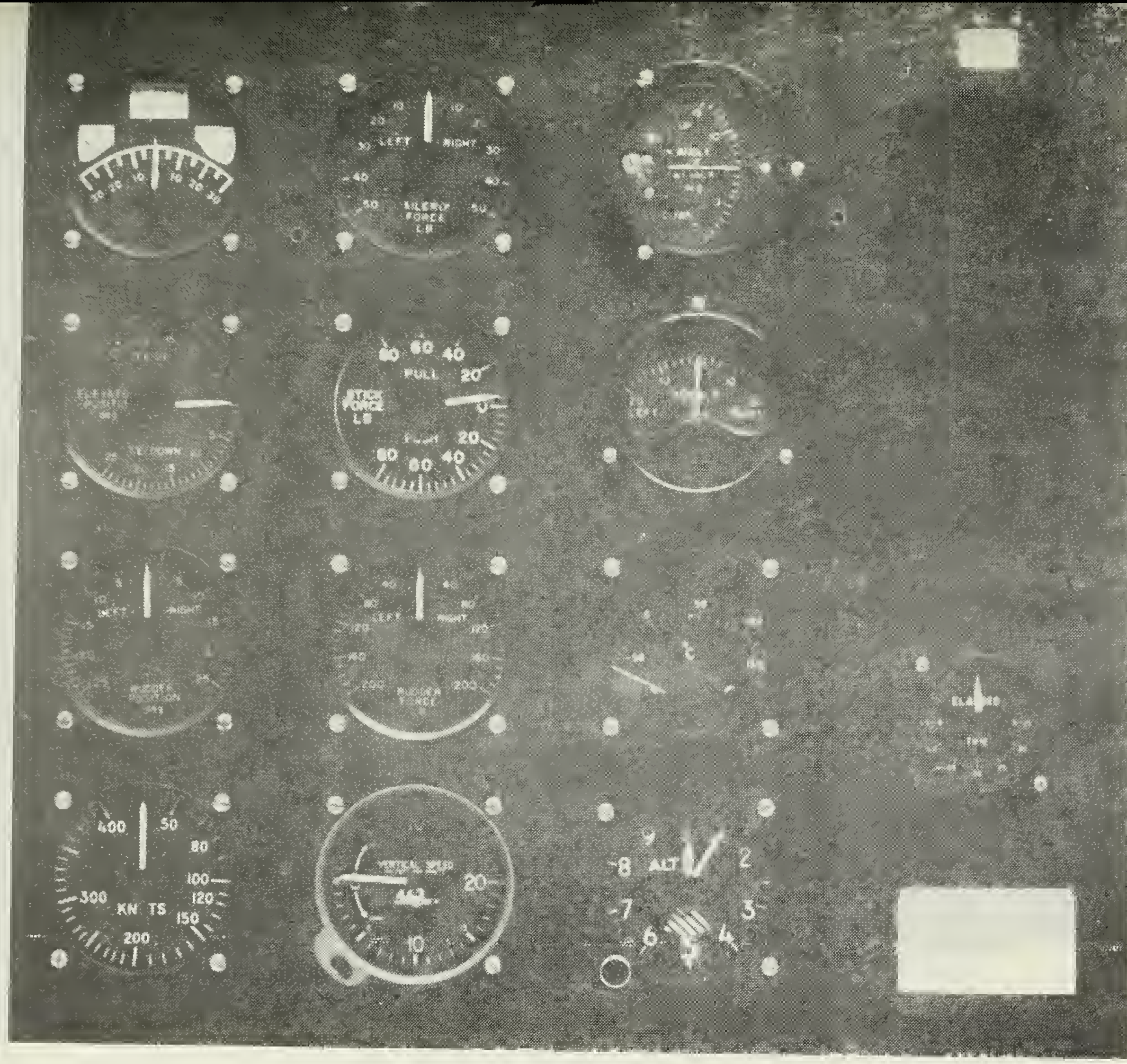


FIGURE 2 PHOTO-PANEL THEATER

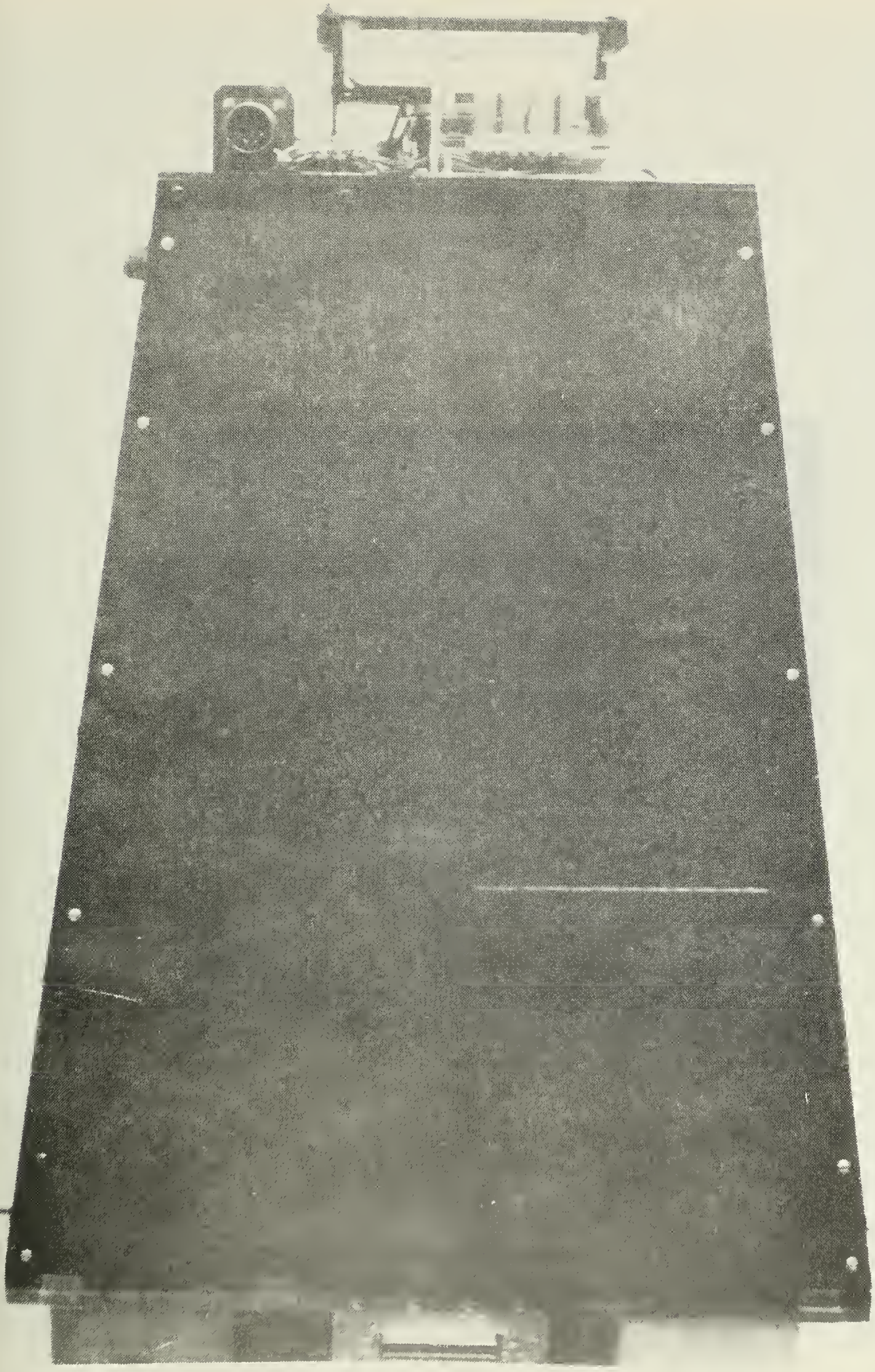


FIGURE 3 PHOTO-PANEL INSTALLATION

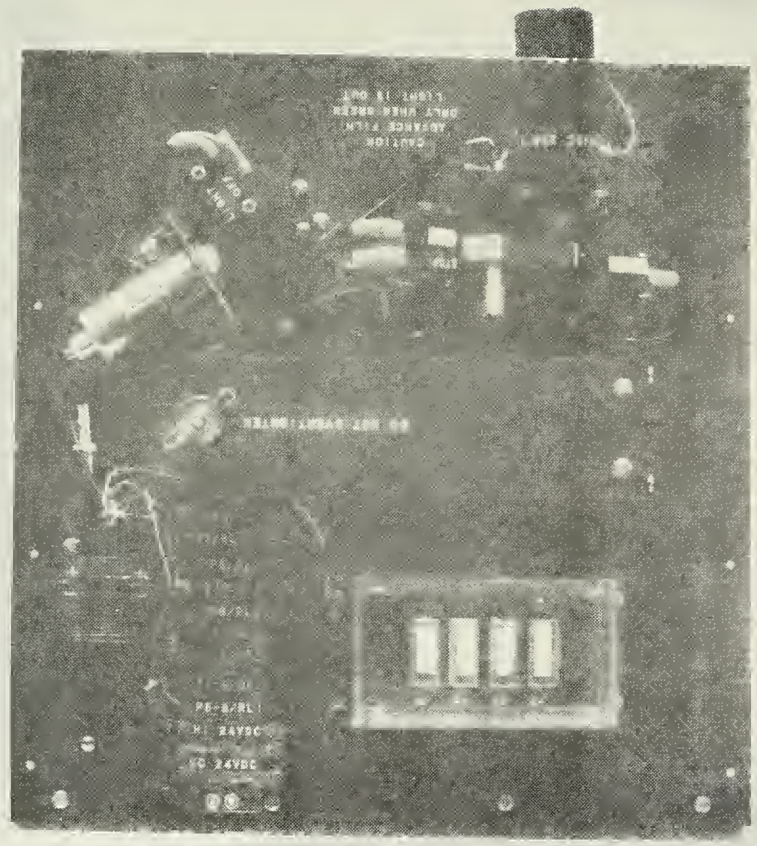


FIGURE 4 CAMERA CONTROL SYSTEM

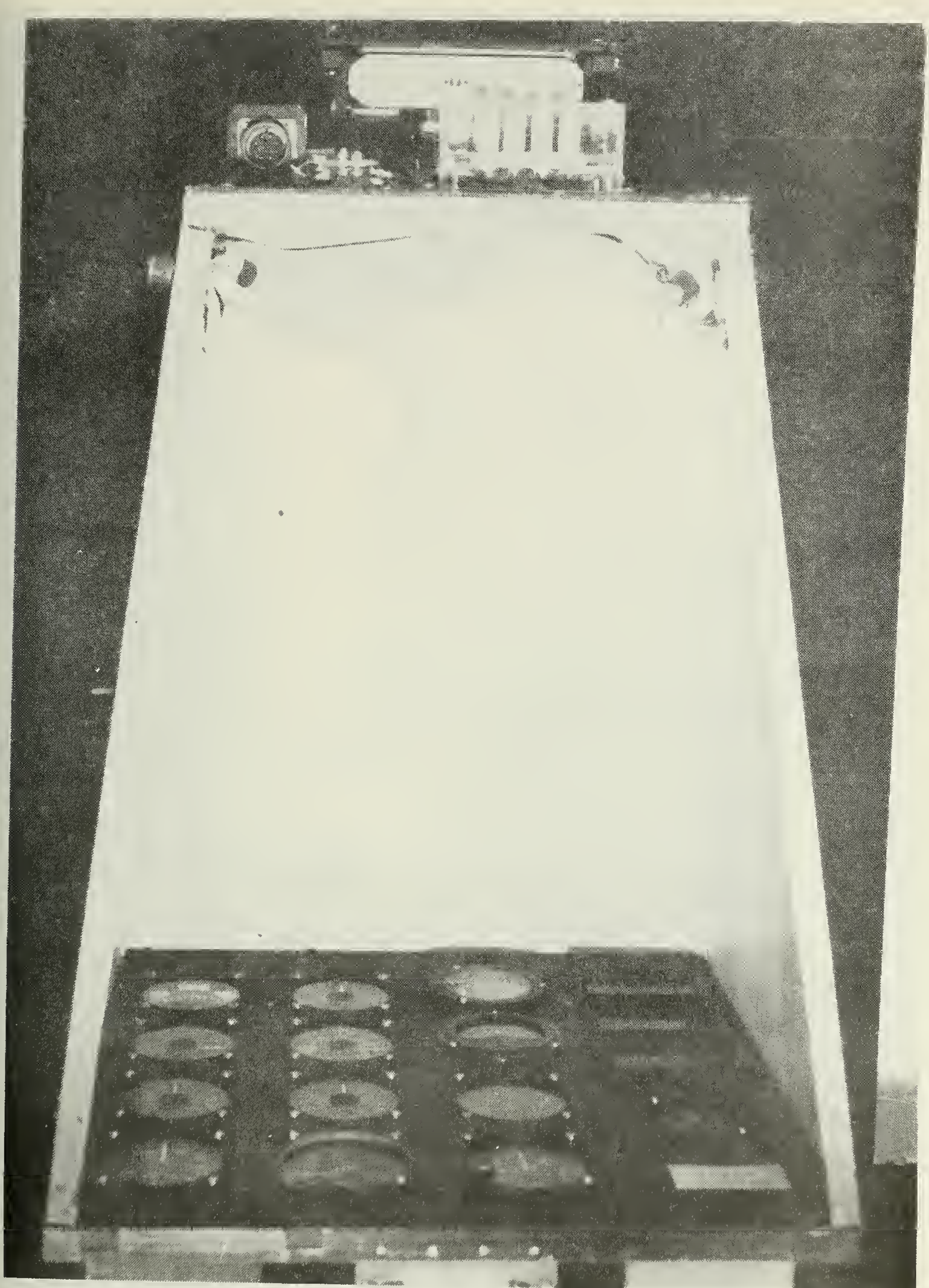


FIGURE 5 OPEN PHOTO-PANEL

FIGURE 6 THREE DEGREE OF FREEDOM LIGHT MOUNT

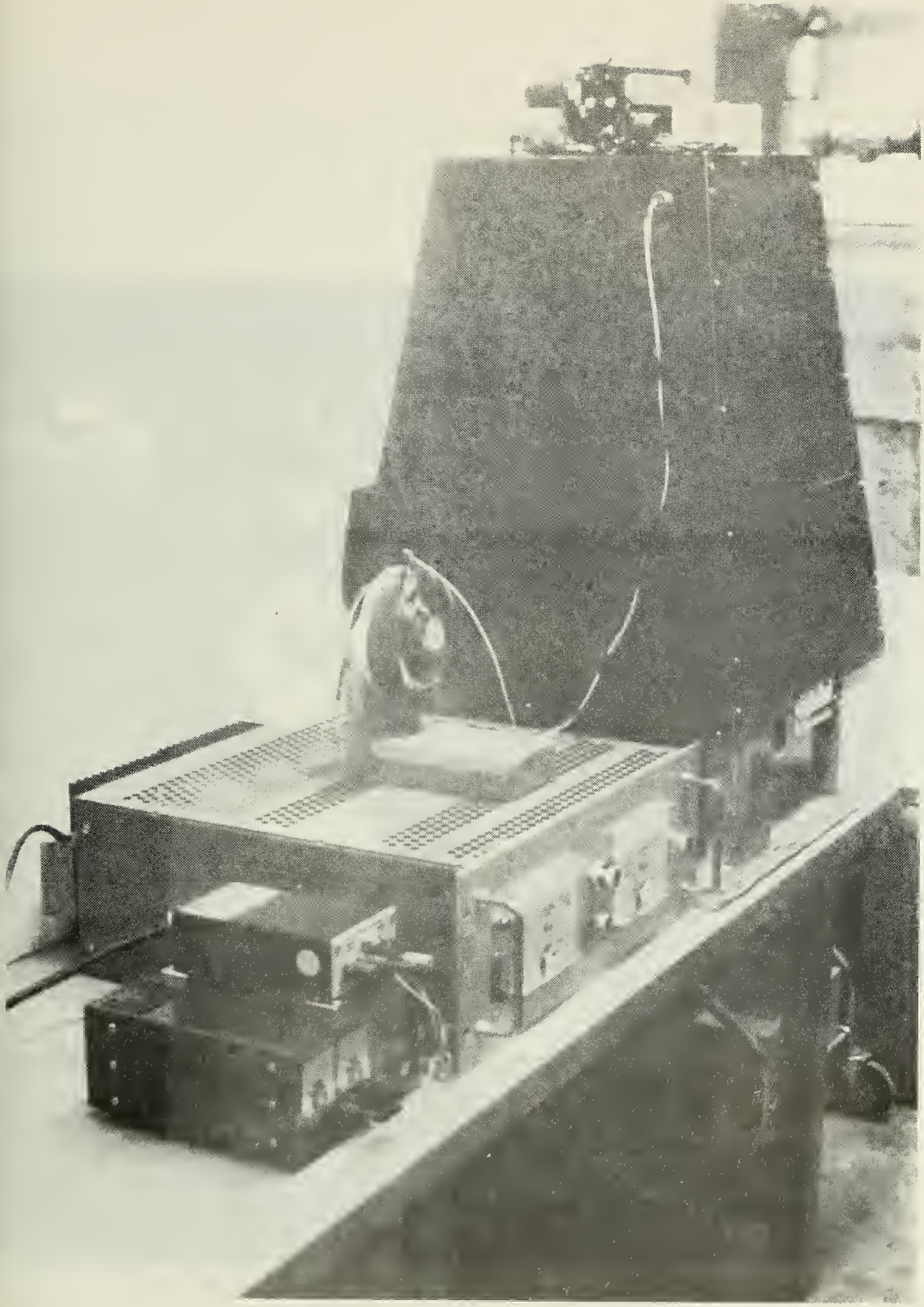


FIGURE 7 RHEOSTAT TESTING APPARATUS

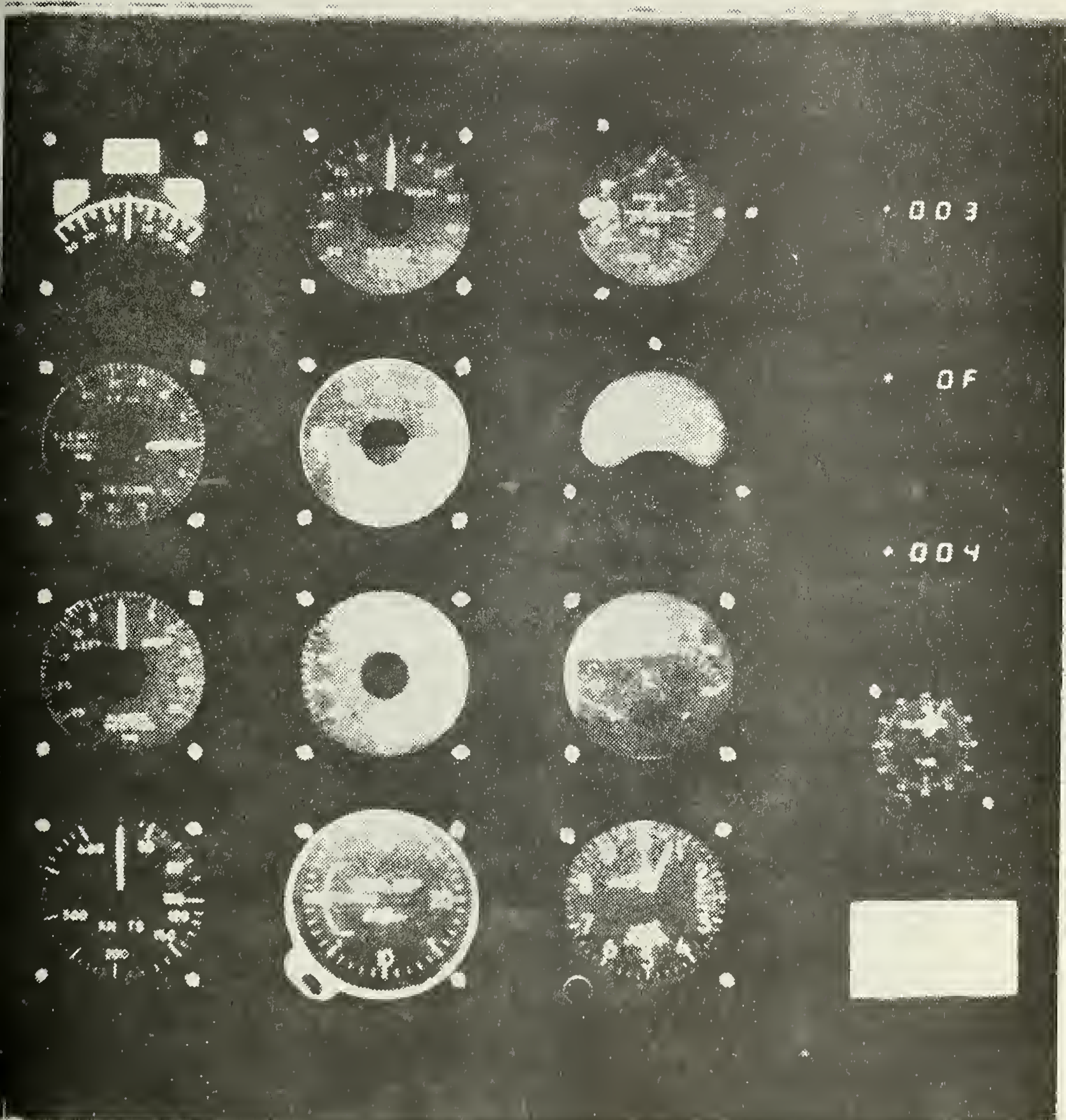


FIGURE 8 SAMPLE PHOTO PANEL OUTPUT

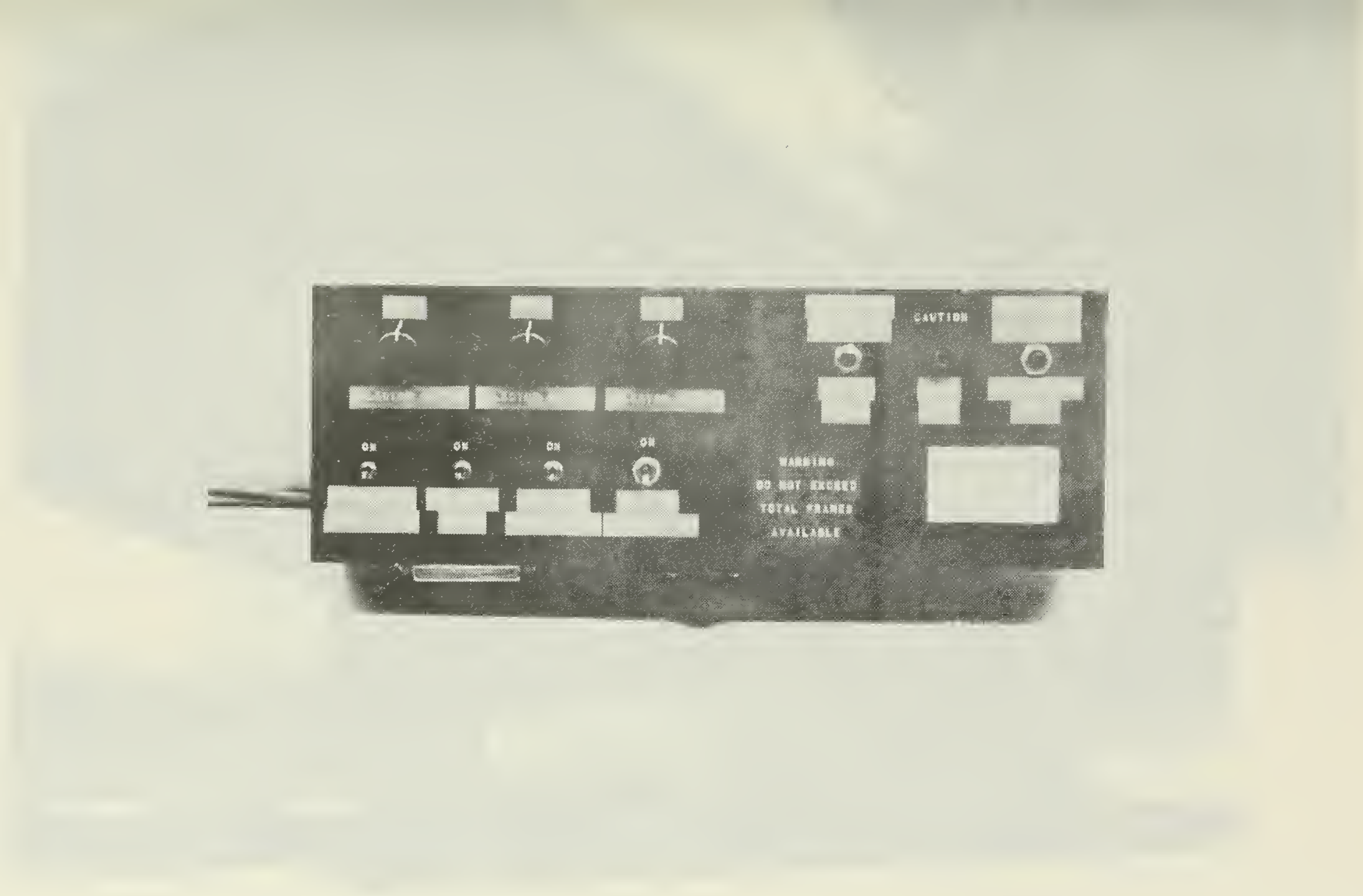


FIGURE 9 CONTROL PANEL INSTALLATION

FIGURE 10 DATEL SYSTEM DIGITAL PANEL-METER

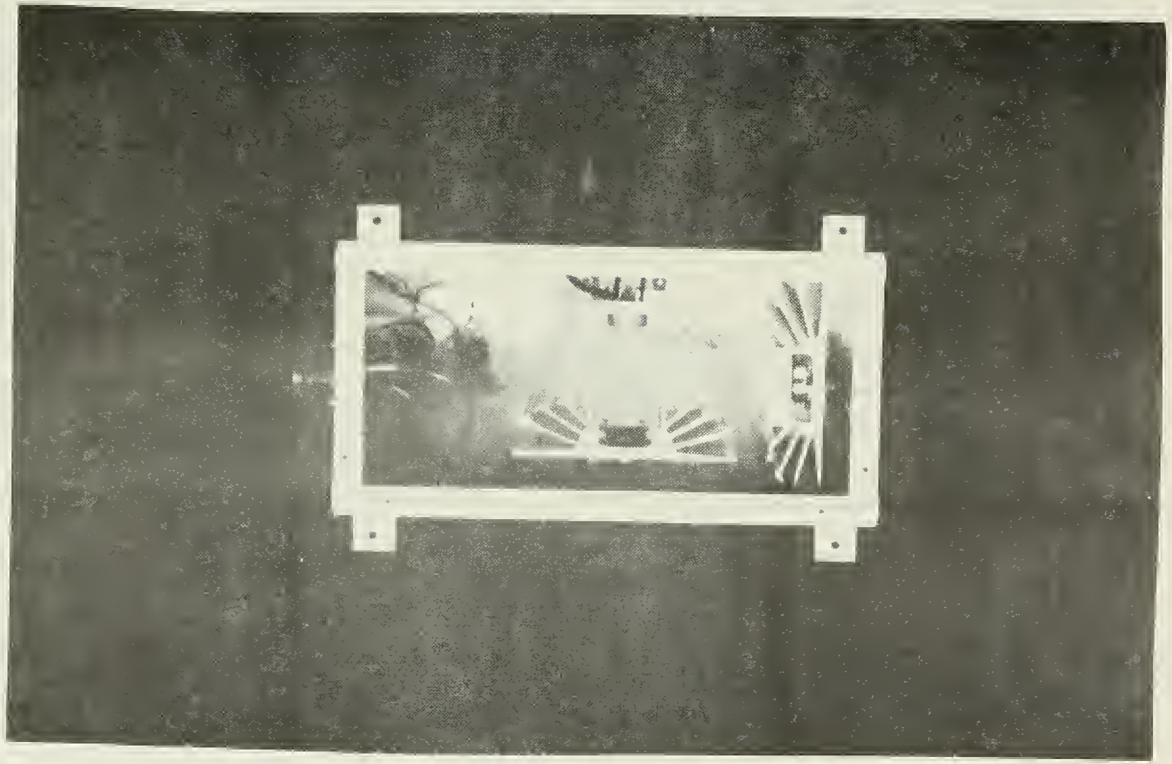
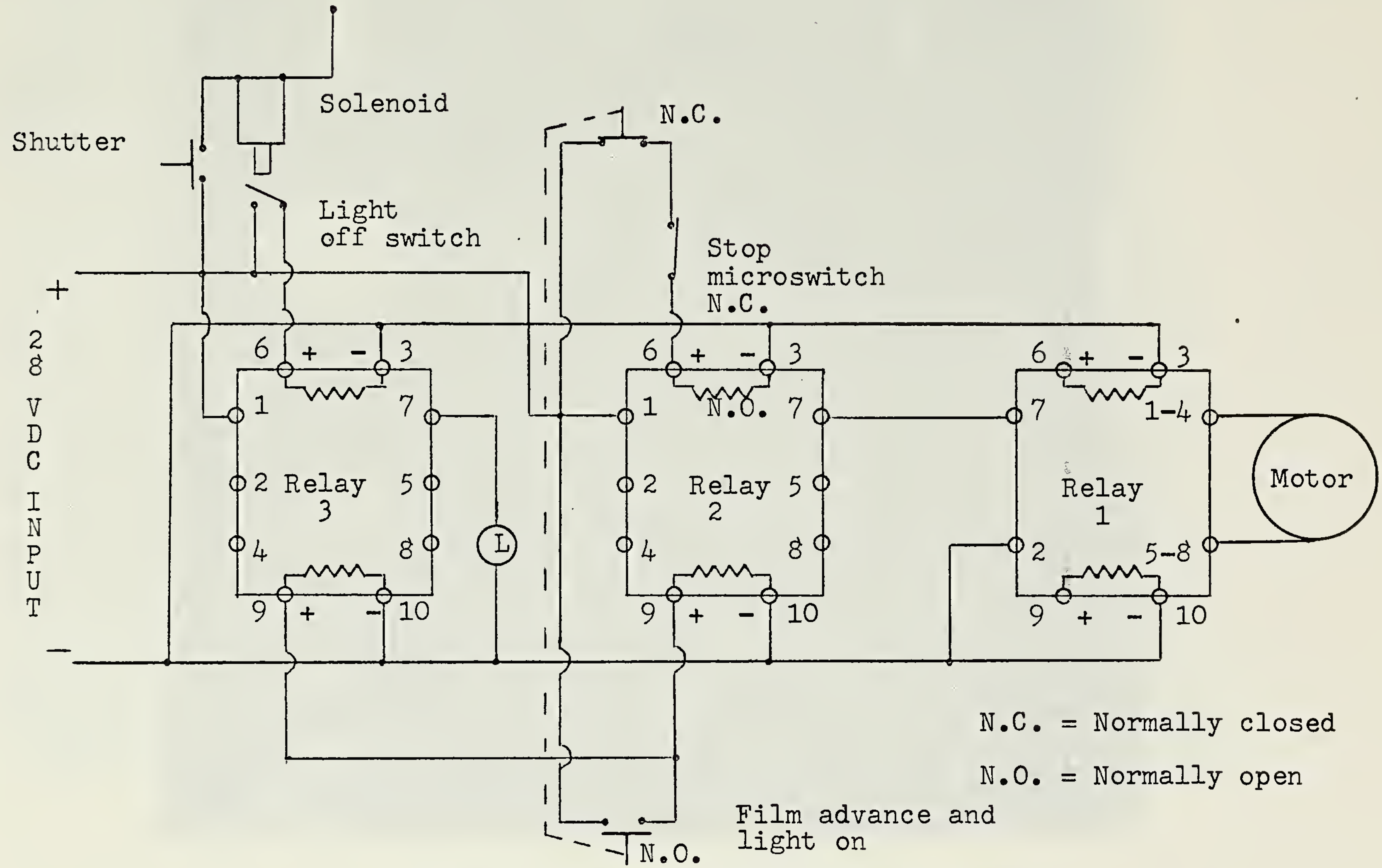


FIGURE 11 DIGITAL PANEL-METER VOLTAGE REGULATOR AND HEAT SINK APPARATUS



N.C. = Normally closed

N.O. = Normally open

FIGURE 12 CAMERA CONTROL SYSTEM SCHEMATIC

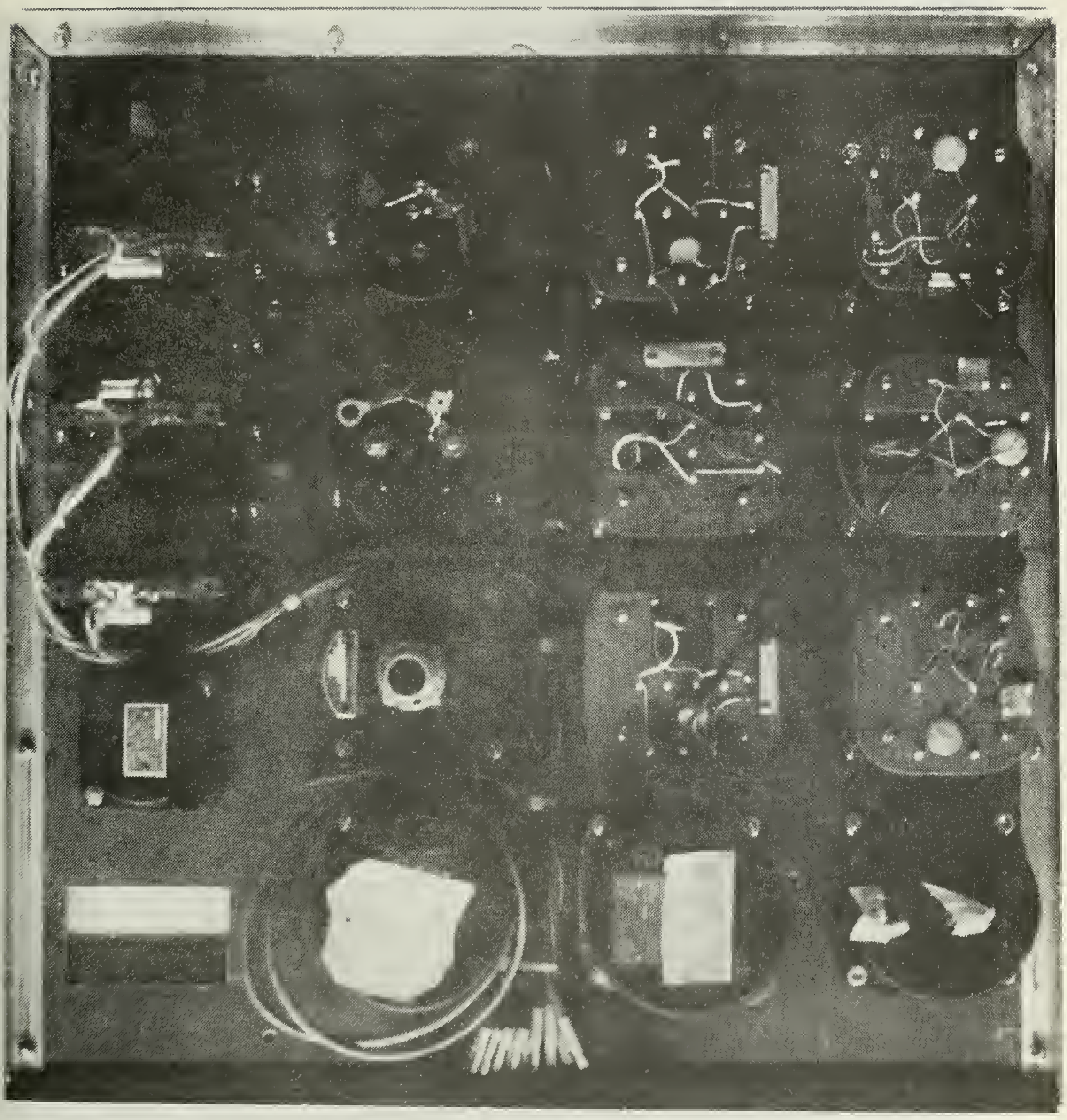


FIGURE 13 PHOTO-PANEL THEATER WIRING VIEW

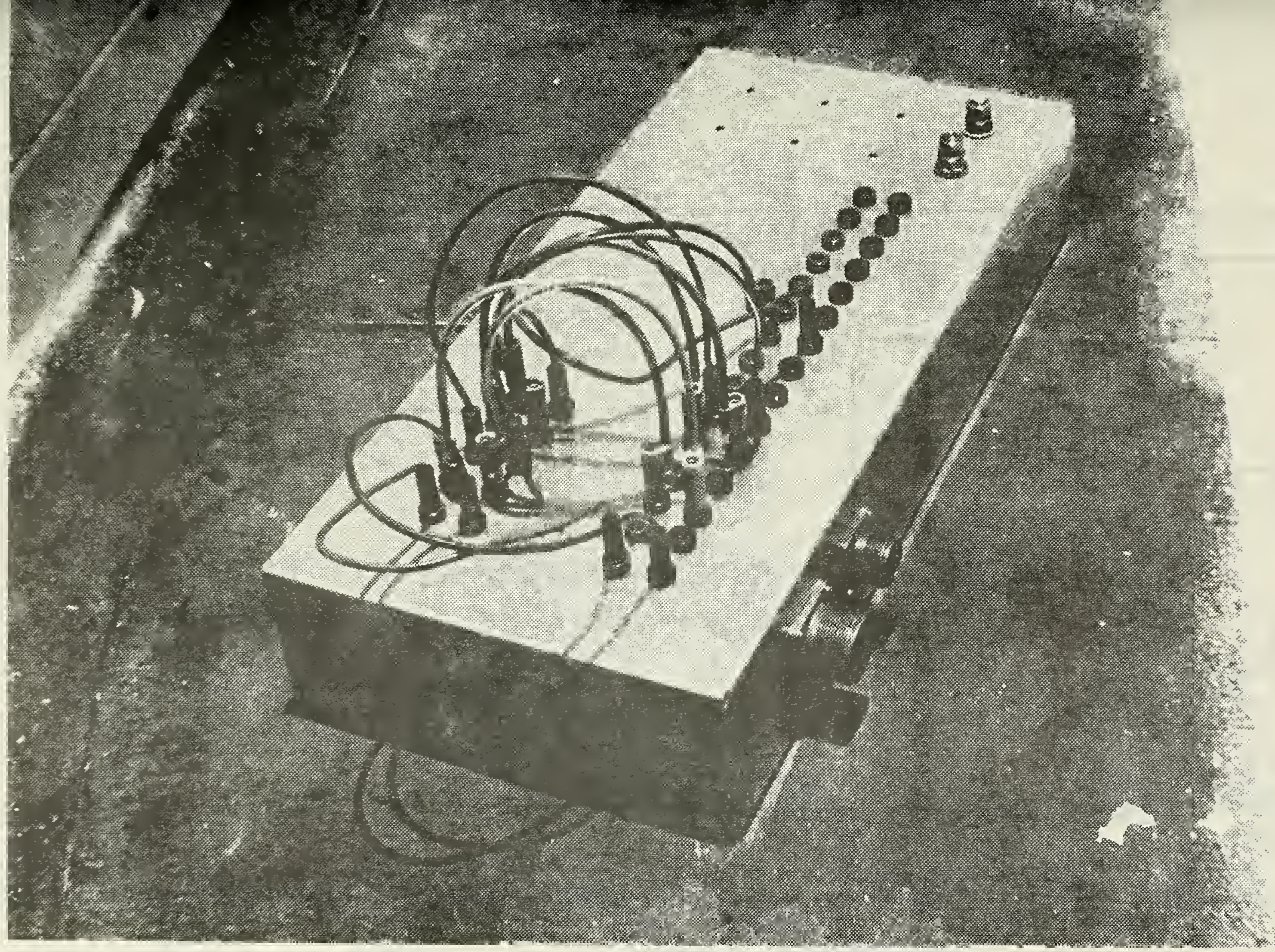


FIGURE 14 JUNCTION BOX

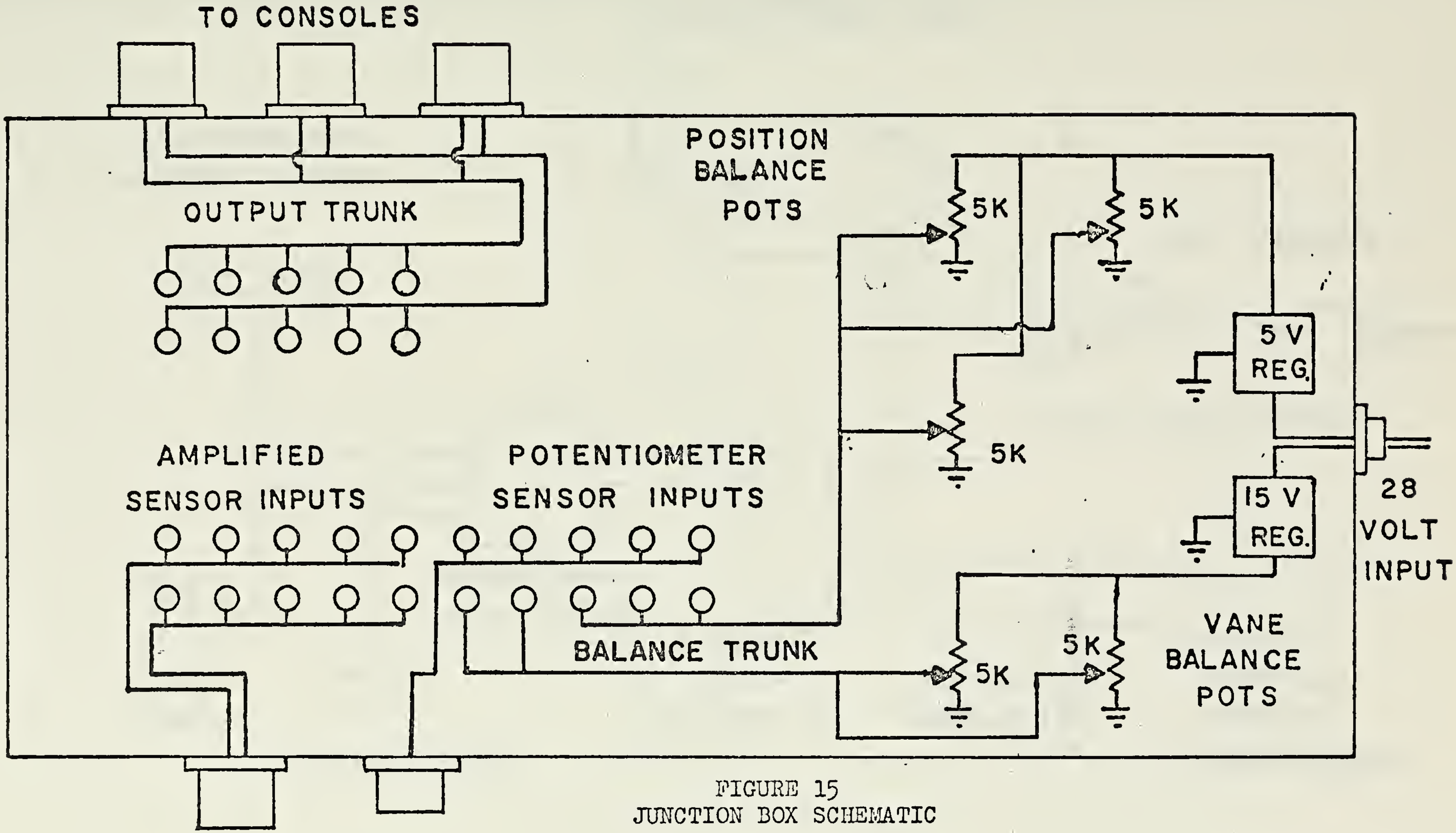


FIGURE 15
JUNCTION BOX SCHEMATIC

FIGURE 16
ELECTRICAL SENSORY INPUTS

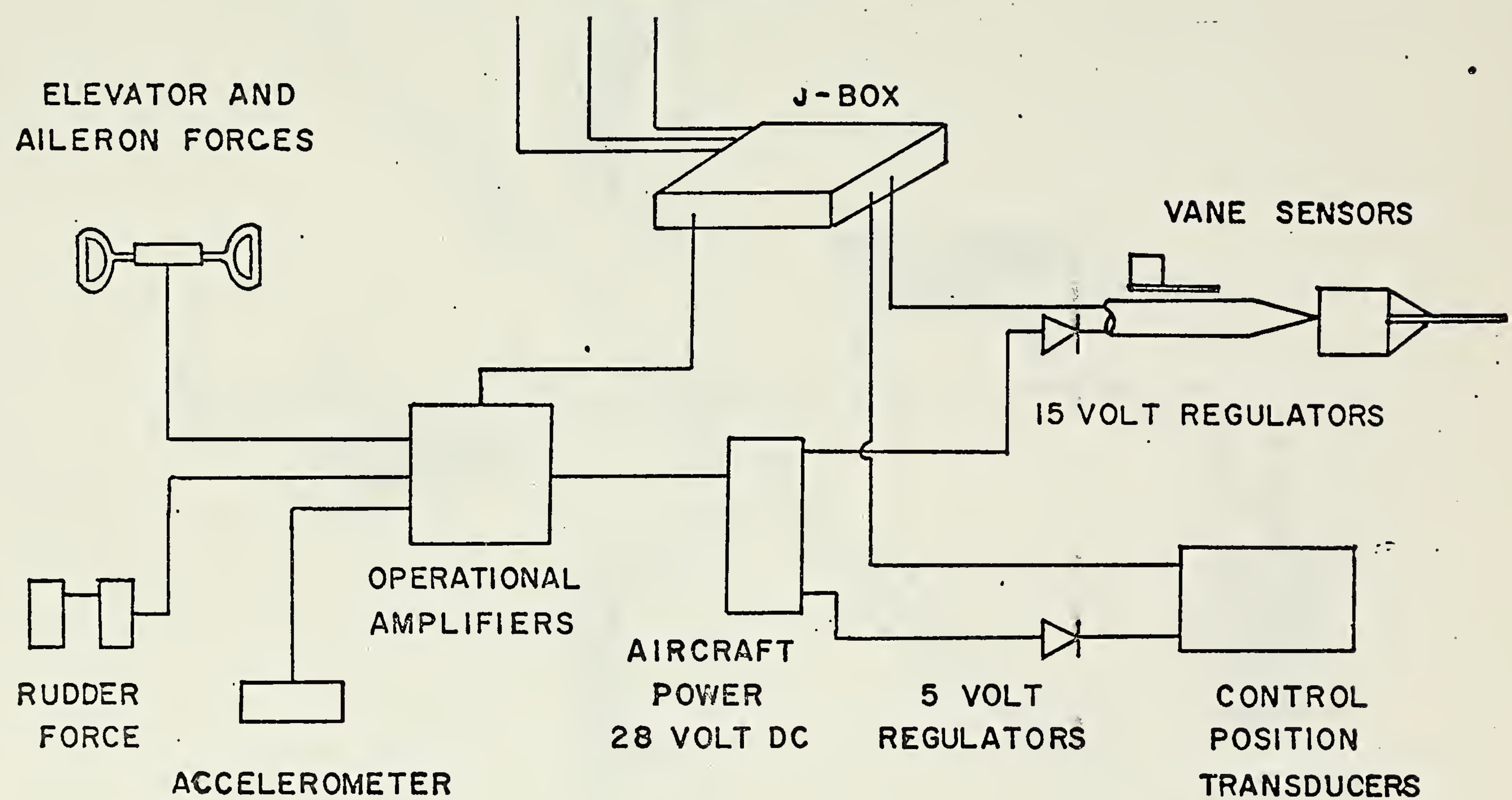


FIGURE 17
AILERON DEFLECTION METER
CALIBRATION
SHUNT RESISTOR 900 OHM

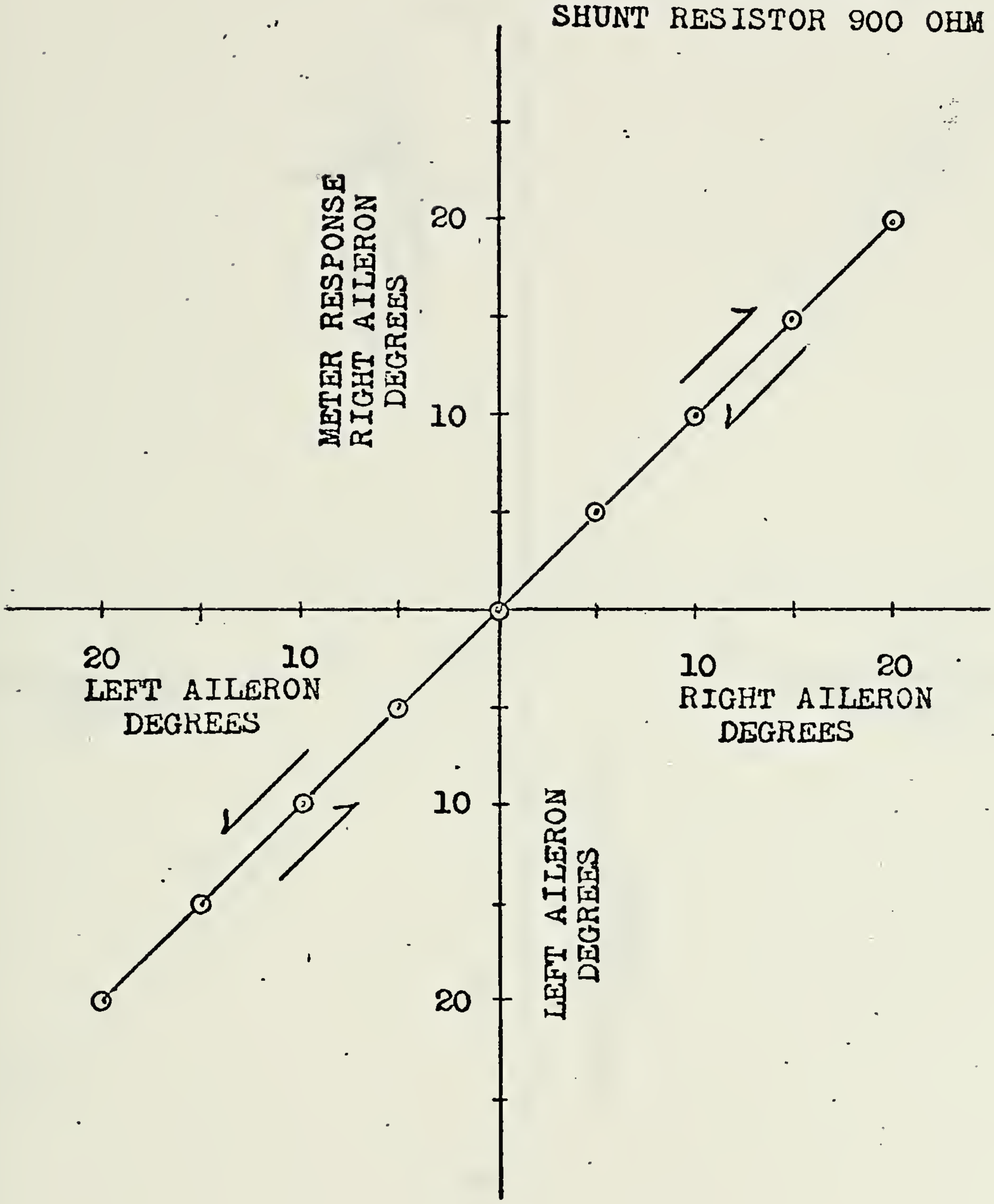


FIGURE 18.
ELEVATOR DEFLECTION METER
CALIBRATION
SHUNT RESISTOR 170 OHM

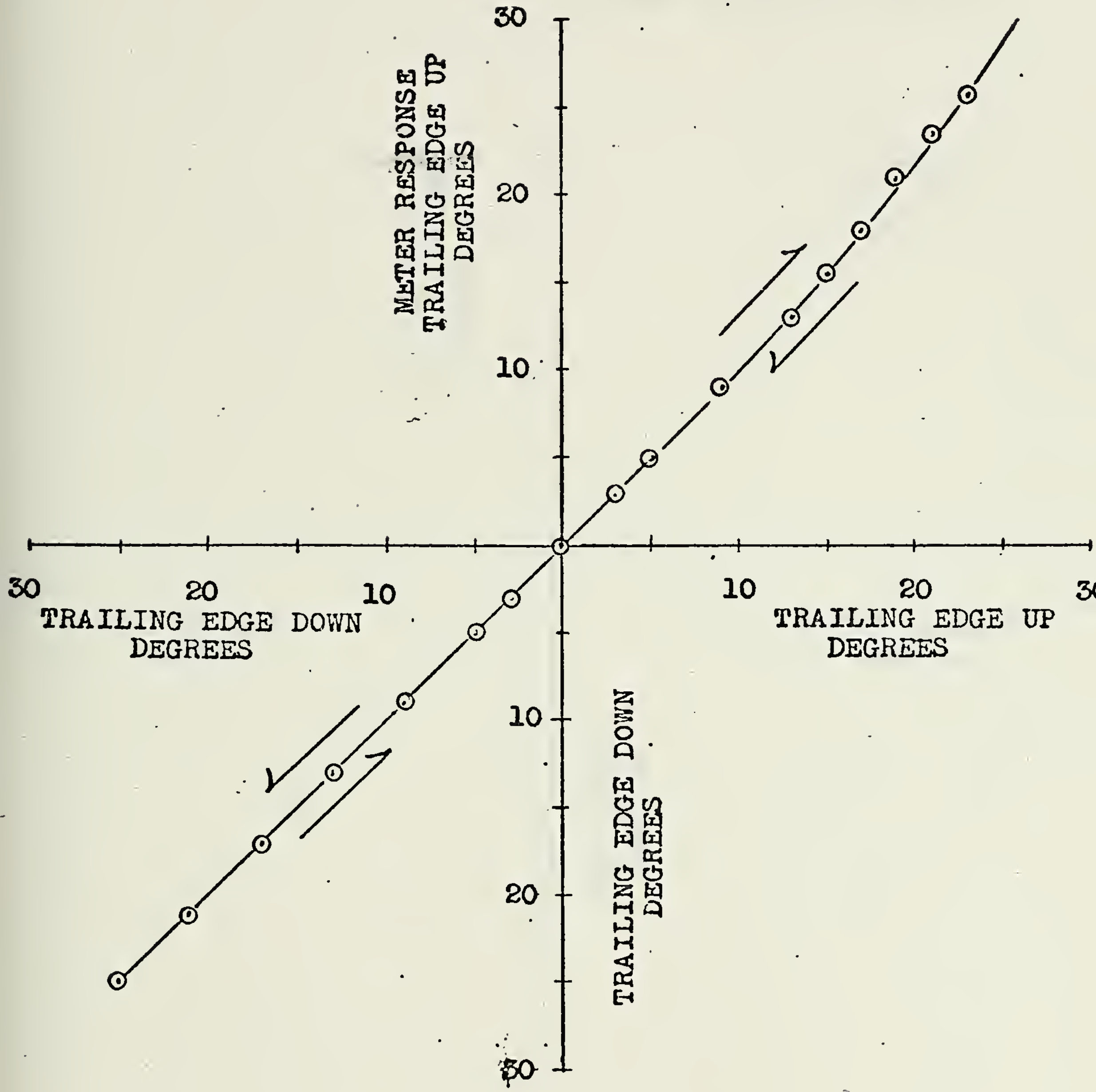


FIGURE 19
RUDDER DEFLECTION METER
CALIBRATION
SHUNT RESISTOR 290 OHM

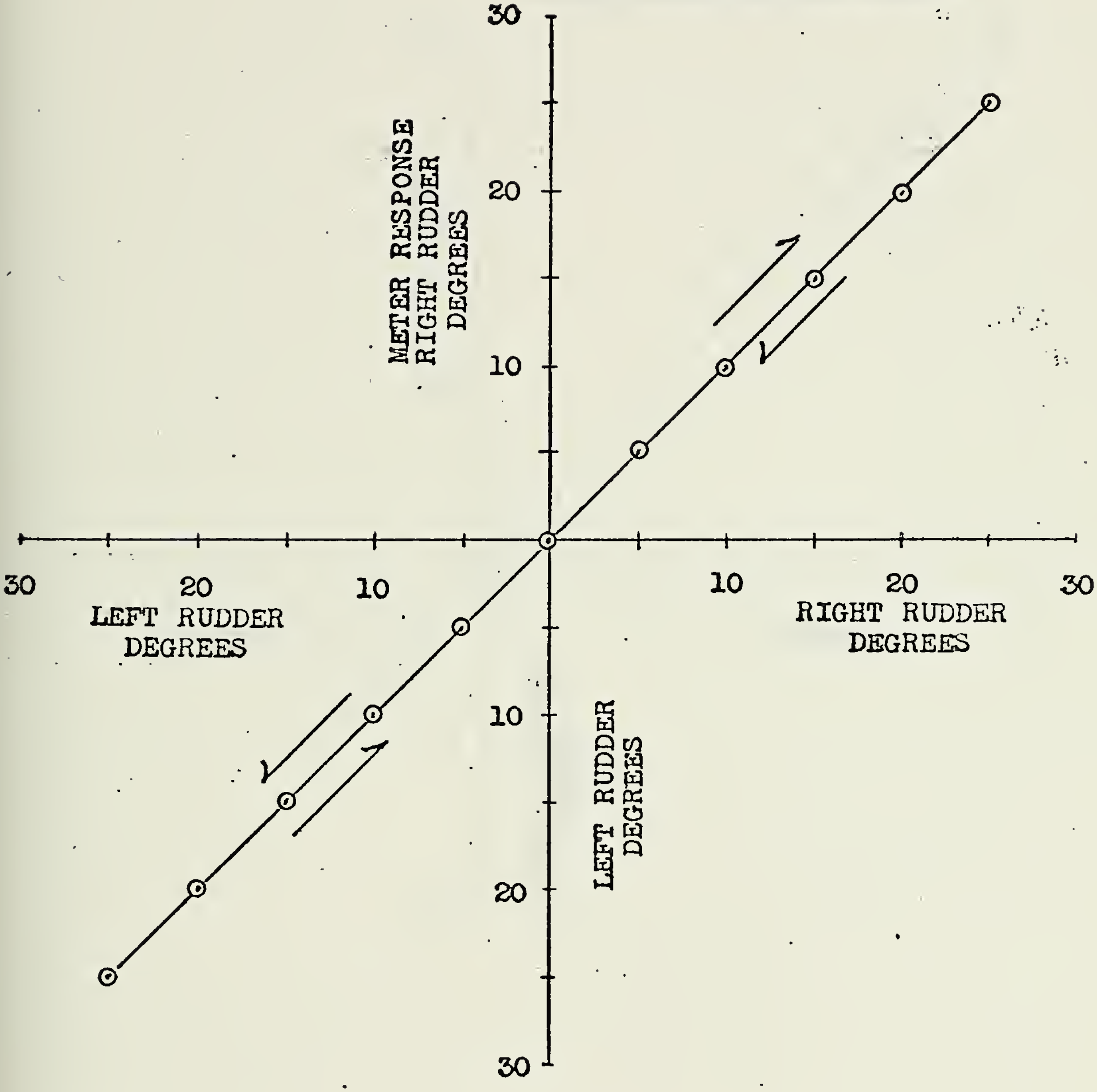


FIGURE 20
ANGLE OF ATTACK METER
CALIBRATION
SHUNT RESISTOR 200 OHM

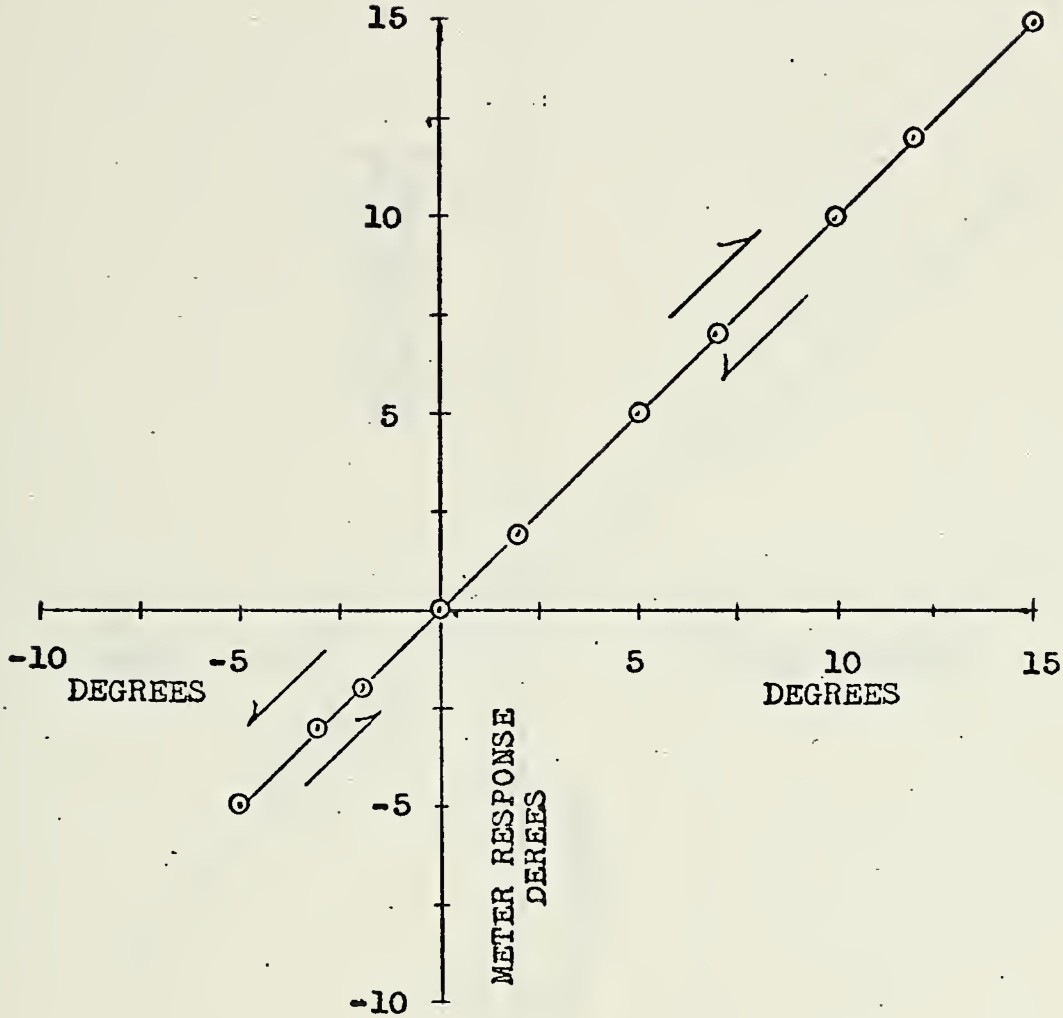


FIGURE 21
SIDESLIP METER CALIBRATION
SHUNT RESISTOR 180 OHM

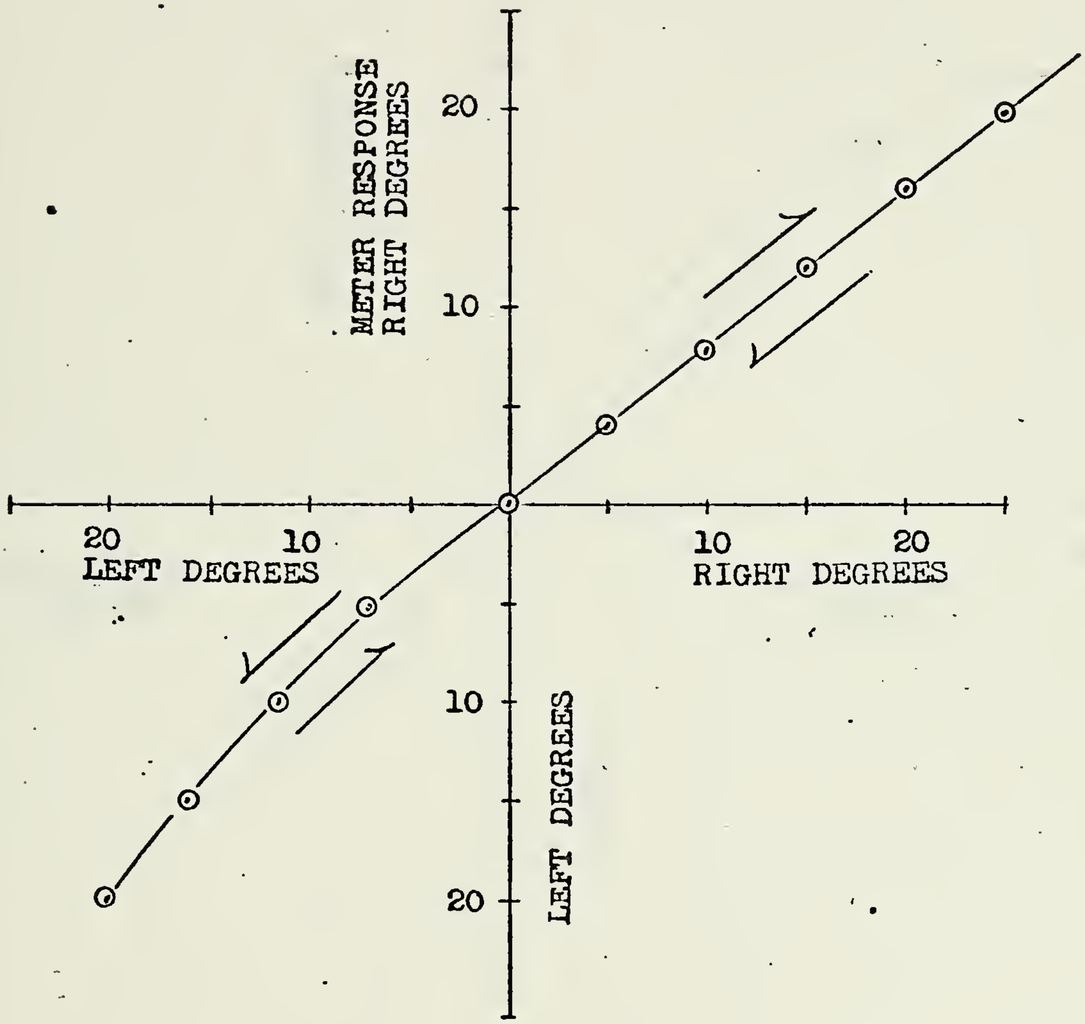


FIGURE 22
AILERON FORCE METER
CALIBRATION
TRIM POTENTIOMETER 340 OHM

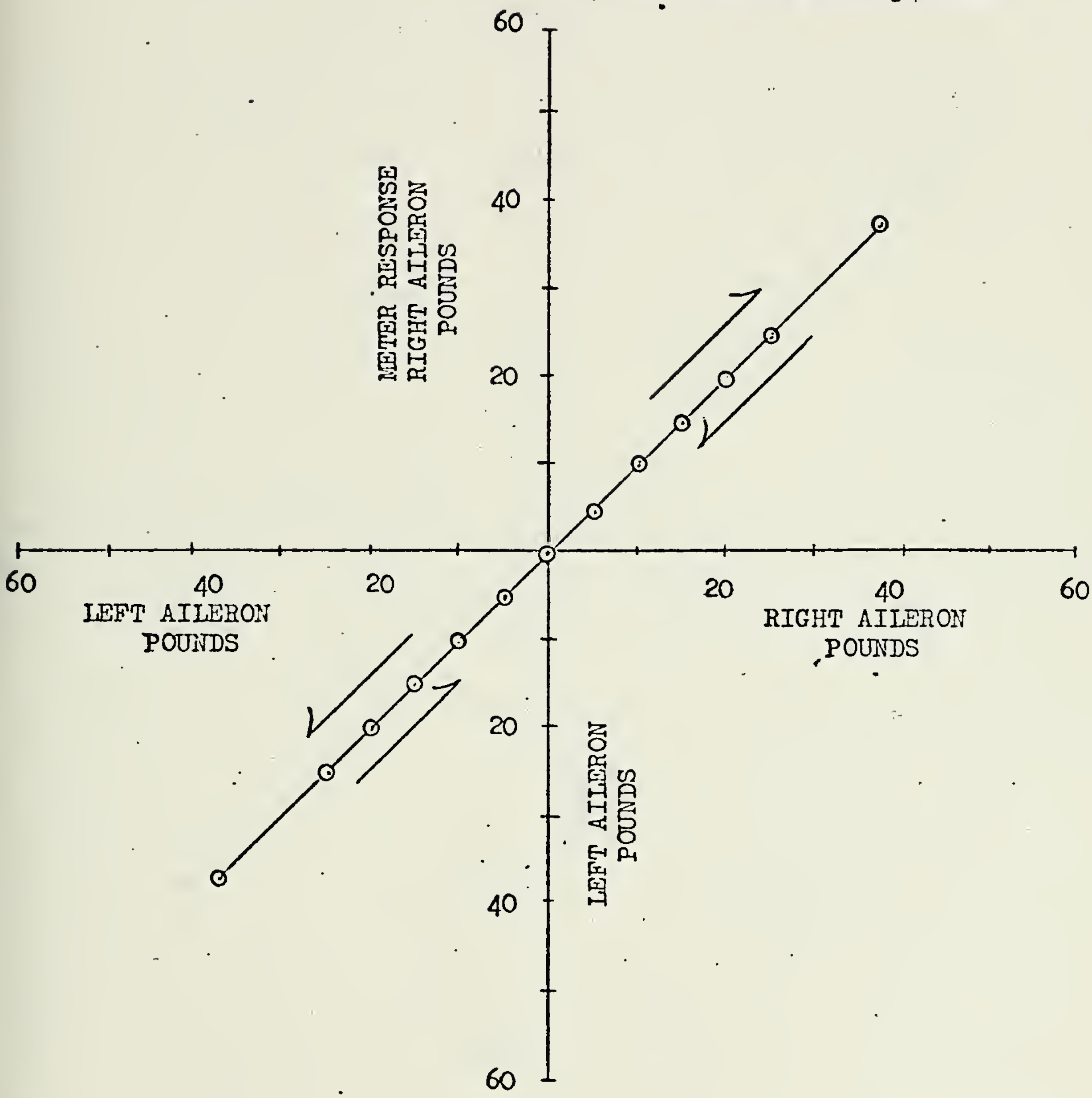


FIGURE 23
ELEVATOR FORCE METER
CALIBRATION
TRIM POTENTIOMETER 1050 OHM

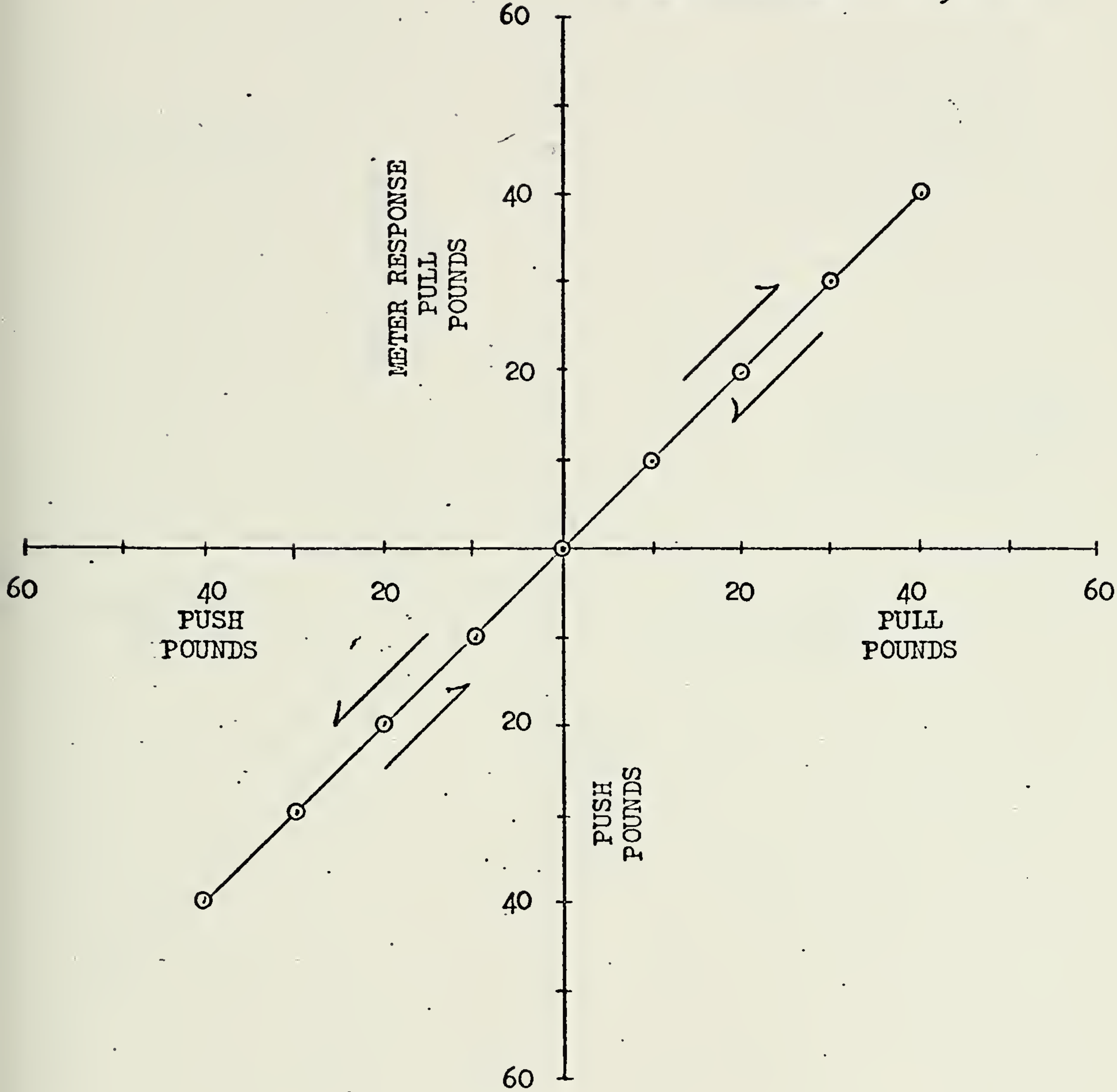
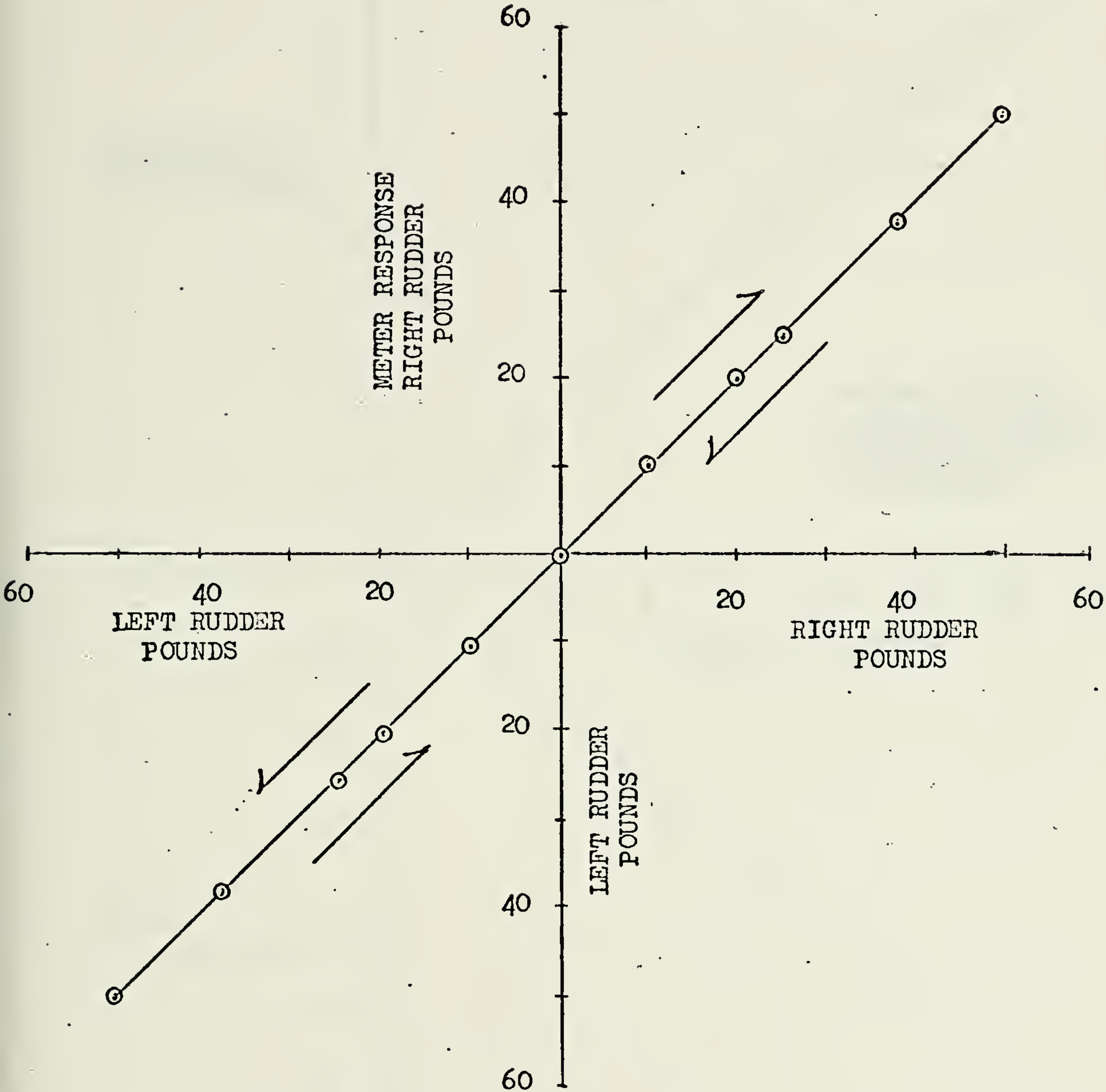


FIGURE 24
RUDDER FORCE METER
CALIBRATION
TRIM POTENTIOMETER 1800 OHM



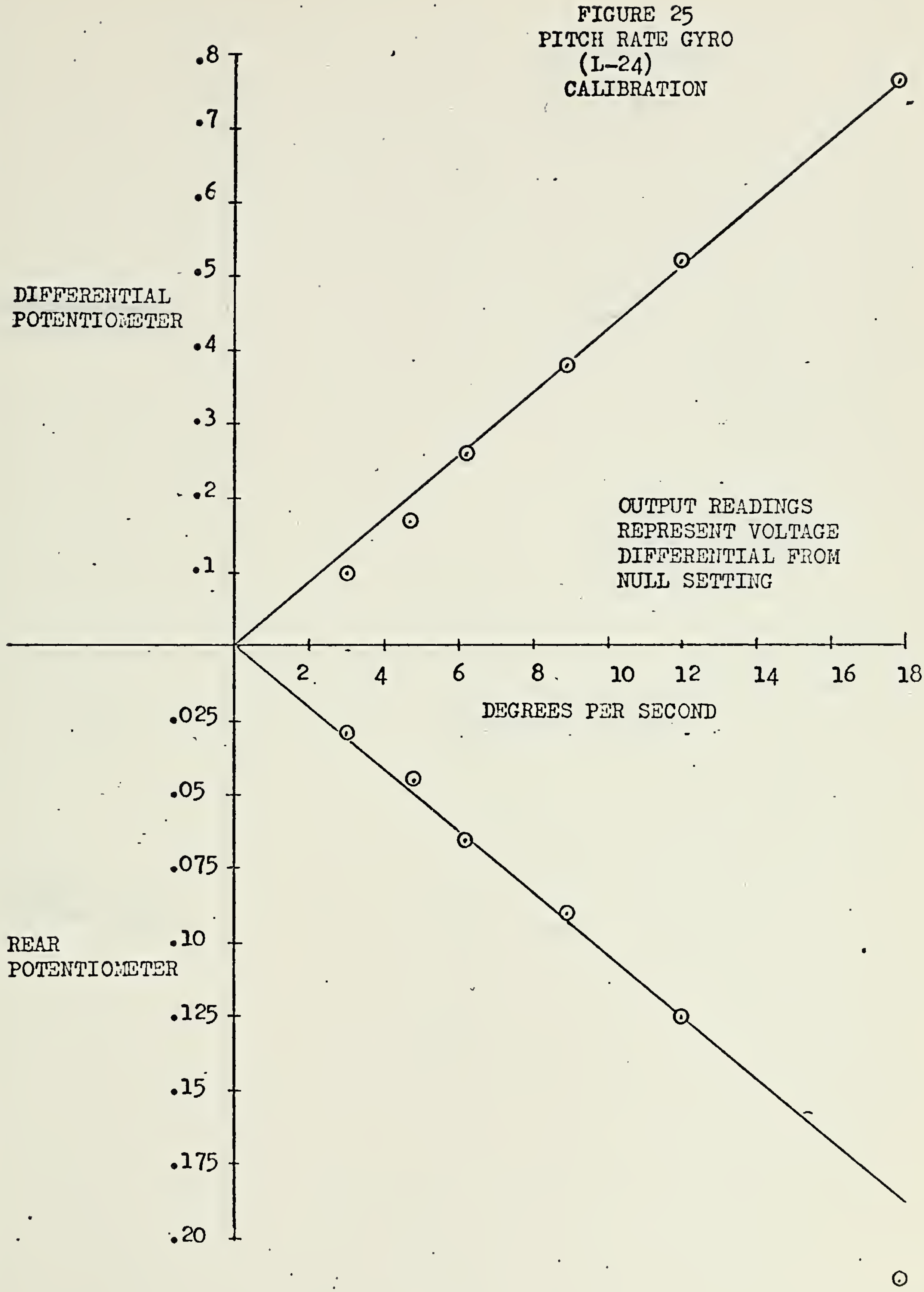


FIGURE 25
PITCH RATE GYRO
(L-24)
CALIBRATION
DIFFERENTIAL
POTENTIOMETER
REAR
POTENTIOMETER
OUTPUT READINGS
REPRESENT VOLTAGE
DIFFERENTIAL FROM
NULL SETTING
DEGREES PER SECOND
.8
.7
.6
.5
.4
.3
.2
.1
.025
.05
.075
.10
.125
.15
.175
.20
2
4
6
8
10
12
14
16
18

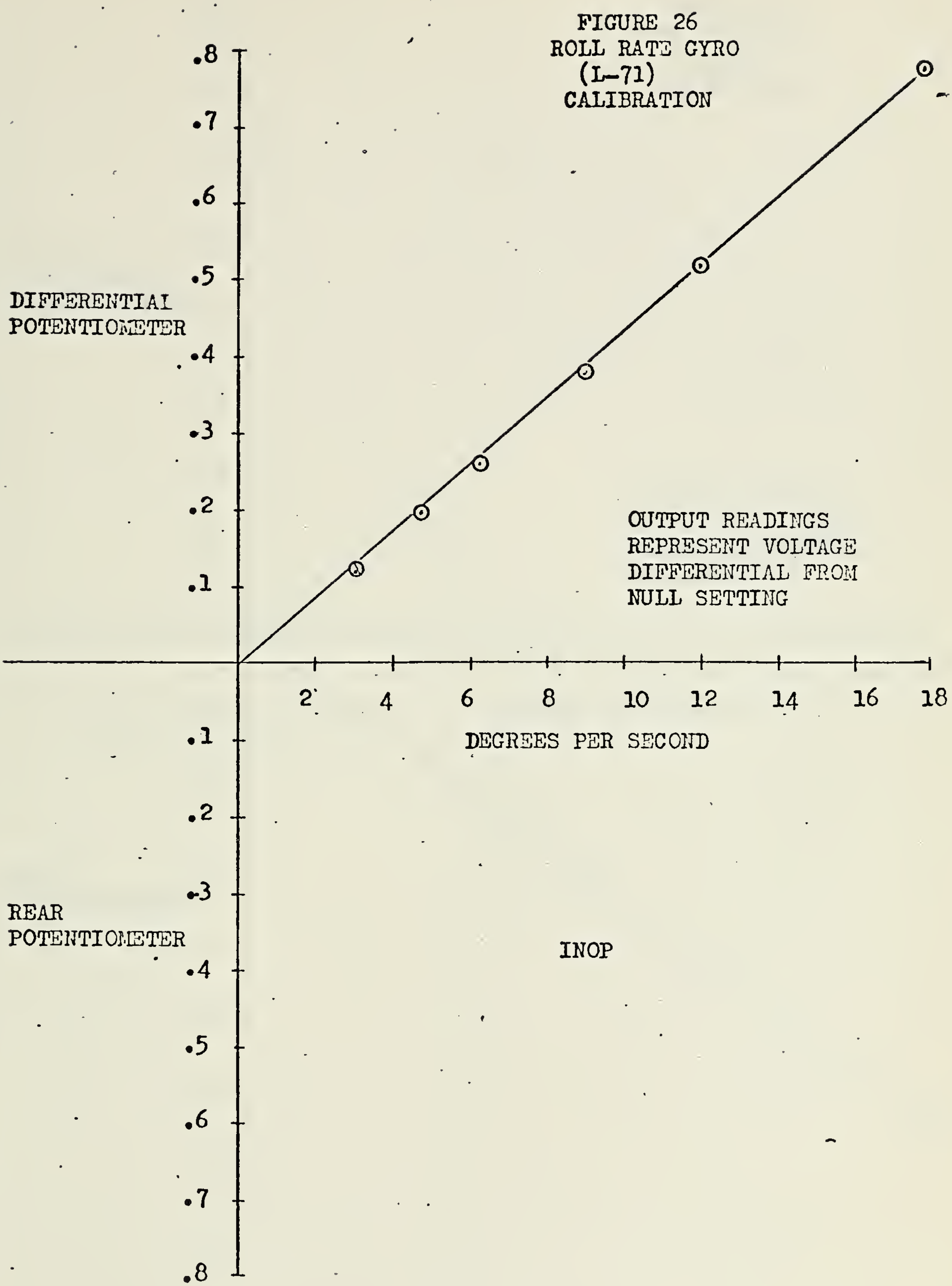


FIGURE 26
ROLL RATE GYRO
(L-71)
CALIBRATION
DIFFERENTIAL POTENTIOMETER
REAR POTENTIOMETER
.8
.7
.6
.5
.4
.3
.2
.1
OUTPUT READINGS REPRESENT VOLTAGE DIFFERENTIAL FROM NULL SETTING
2
4
6
8
10
12
14
16
18
DEGREES PER SECOND
INOP

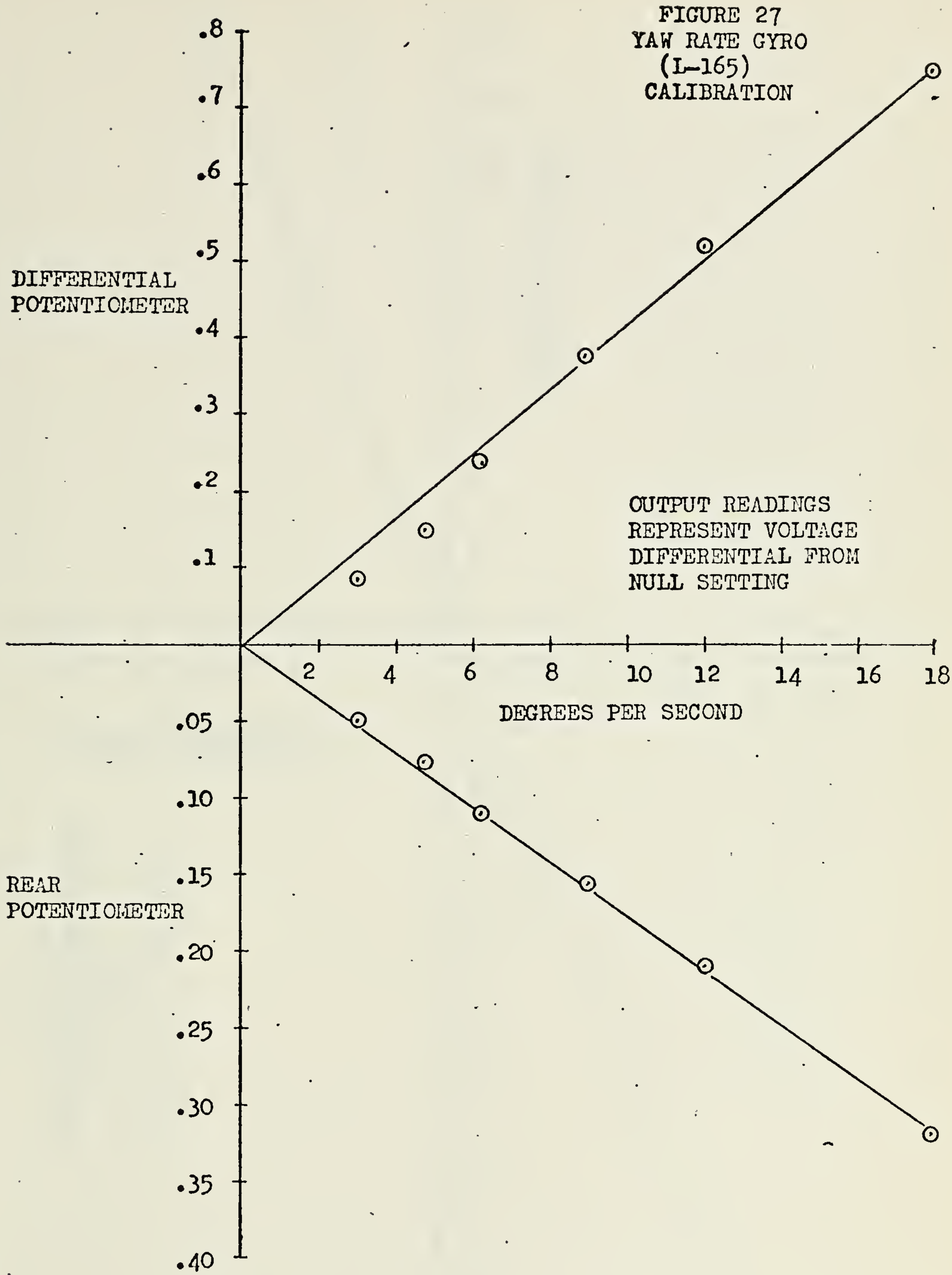


FIGURE 27
YAW RATE GYRO
(L-165)
CALIBRATION
DIFFERENTIAL
POTENTIOMETER
REAR
POTENTIOMETER
OUTPUT READINGS
REPRESENT VOLTAGE
DIFFERENTIAL FROM
NULL SETTING
DEGREES PER SECOND
.8
.7
.6
.5
.4
.3
.2
.1
.05
.10
.15
.20
.25
.30
.35
.40
2
4
6
8
10
12
14
16
18

FIGURE 28
DIFFERENTIAL POTENTIOMETER
OUTPUT
VS
ANGULAR VELOCITY INPUT

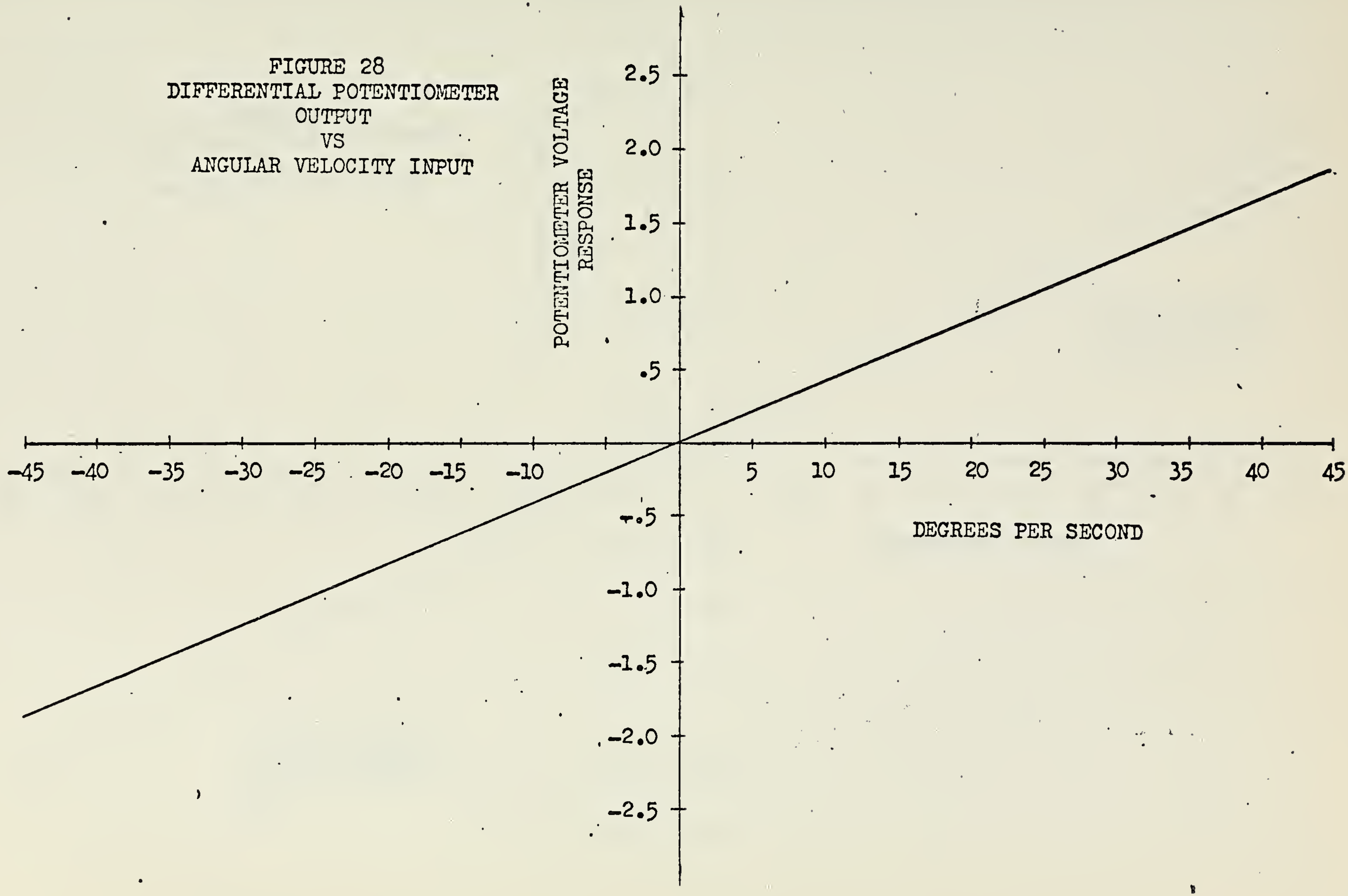
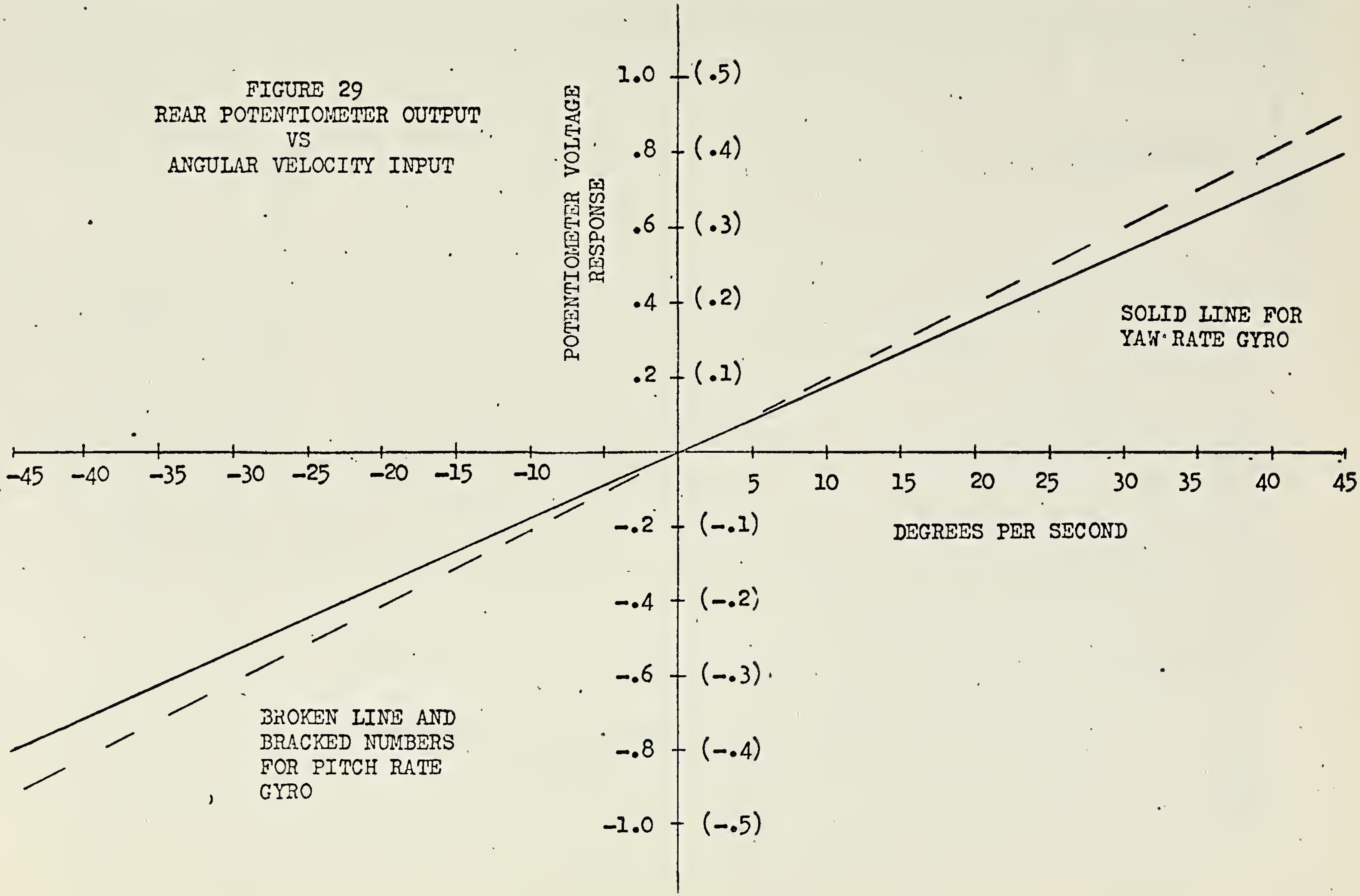
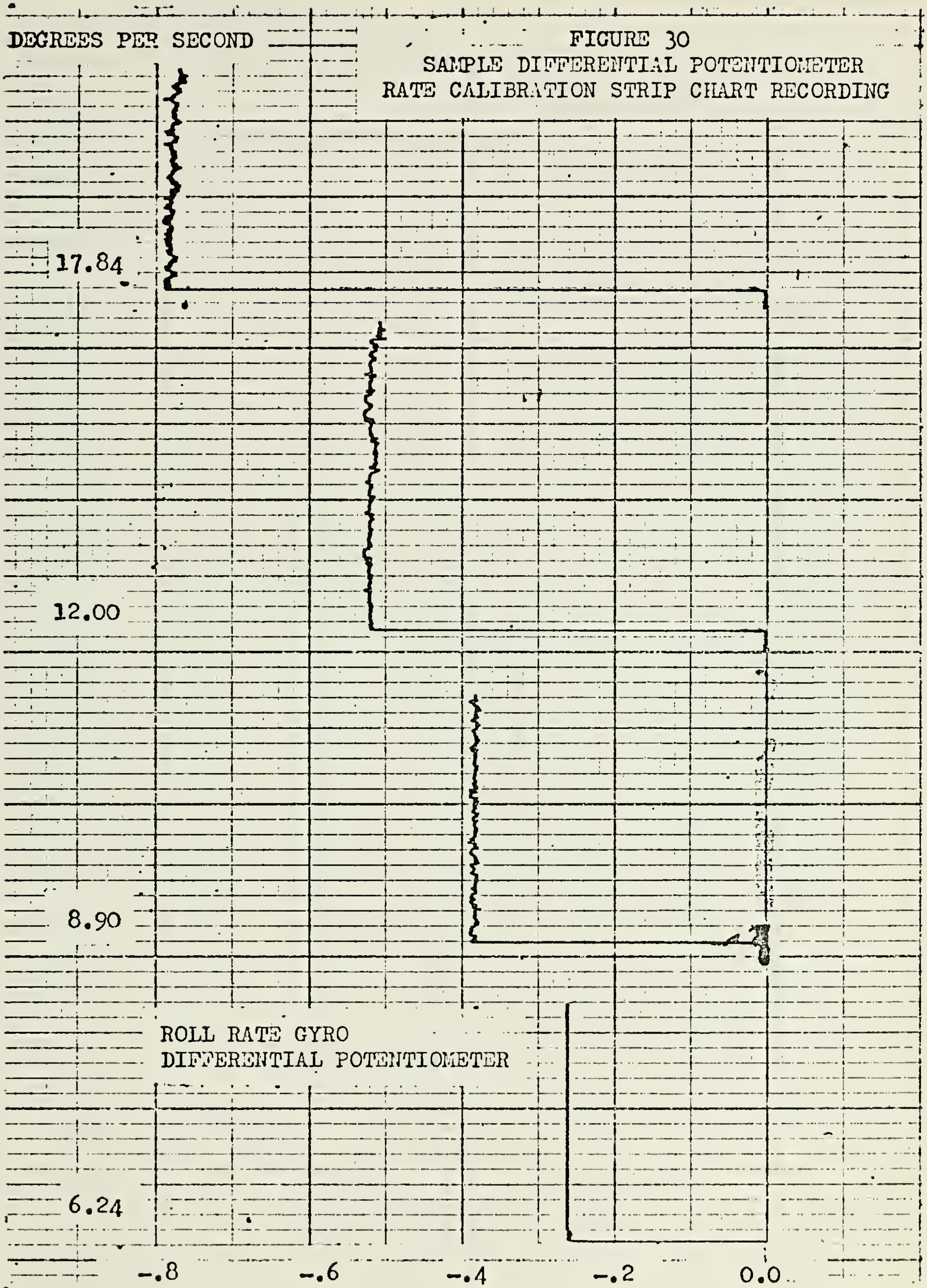


FIGURE 29
REAR POTENTIOMETER OUTPUT
VS
ANGULAR VELOCITY INPUT




DEGREES PER SECOND
FIGURE 30
SAMPLE DIFFERENTIAL POTENTIOMETER
RATE CALIBRATION STRIP CHART RECORDING
17.84
12.00
8.90
ROLL RATE GYRO
DIFFERENTIAL POTENTIOMETER
6.24
-.8
-.6
-.4
-.2
0.0

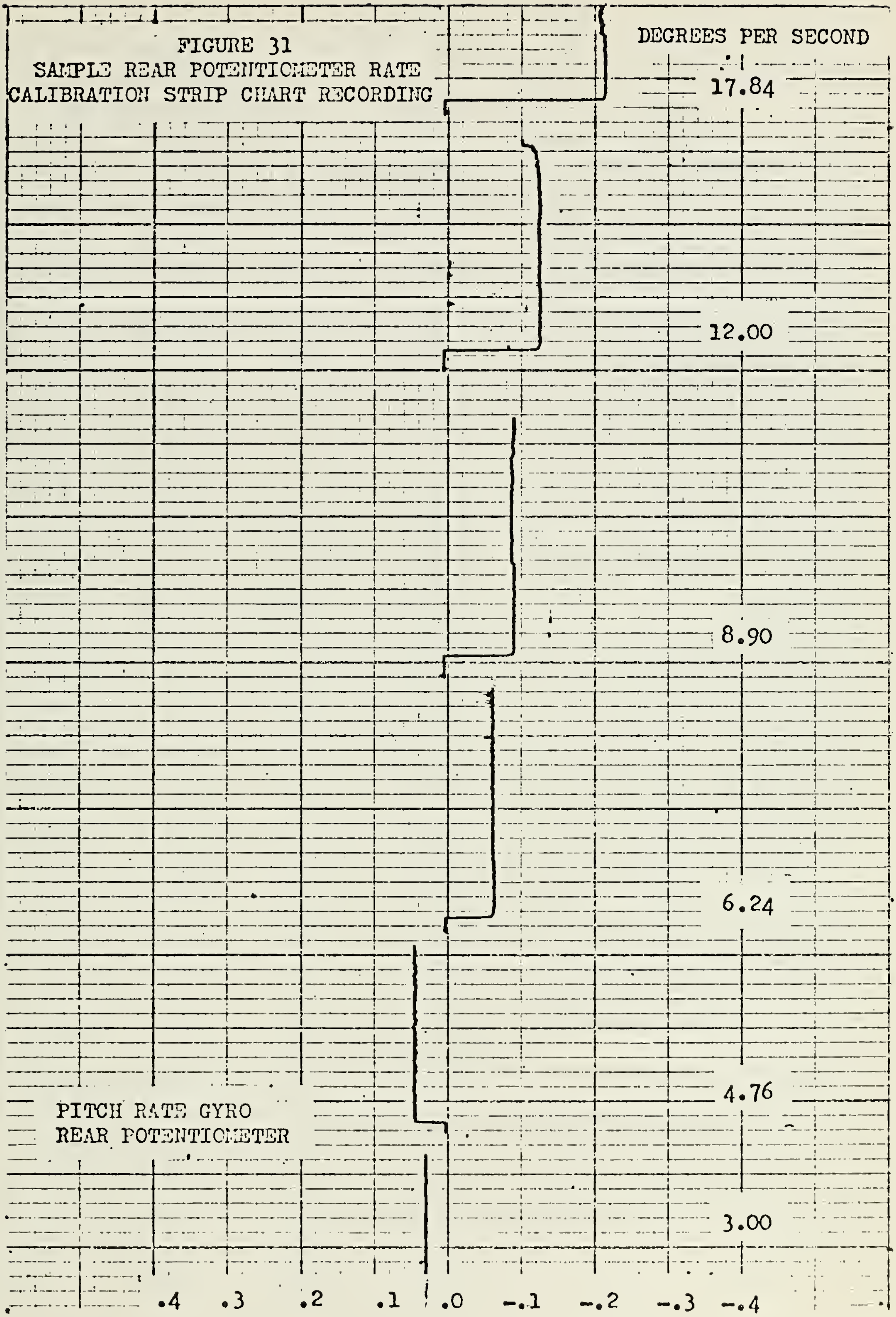


FIGURE 31
SAMPLE REAR POTENTIOMETER RATE
CALIBRATION STRIP CHART RECORDING
DEGREES PER SECOND
17.84
12.00
8.90
6.24
4.76
3.00
PITCH RATE GYRO
REAR POTENTIOMETER
.4
.3
.2
.1
.0
-.1
-.2
-.3
-.4

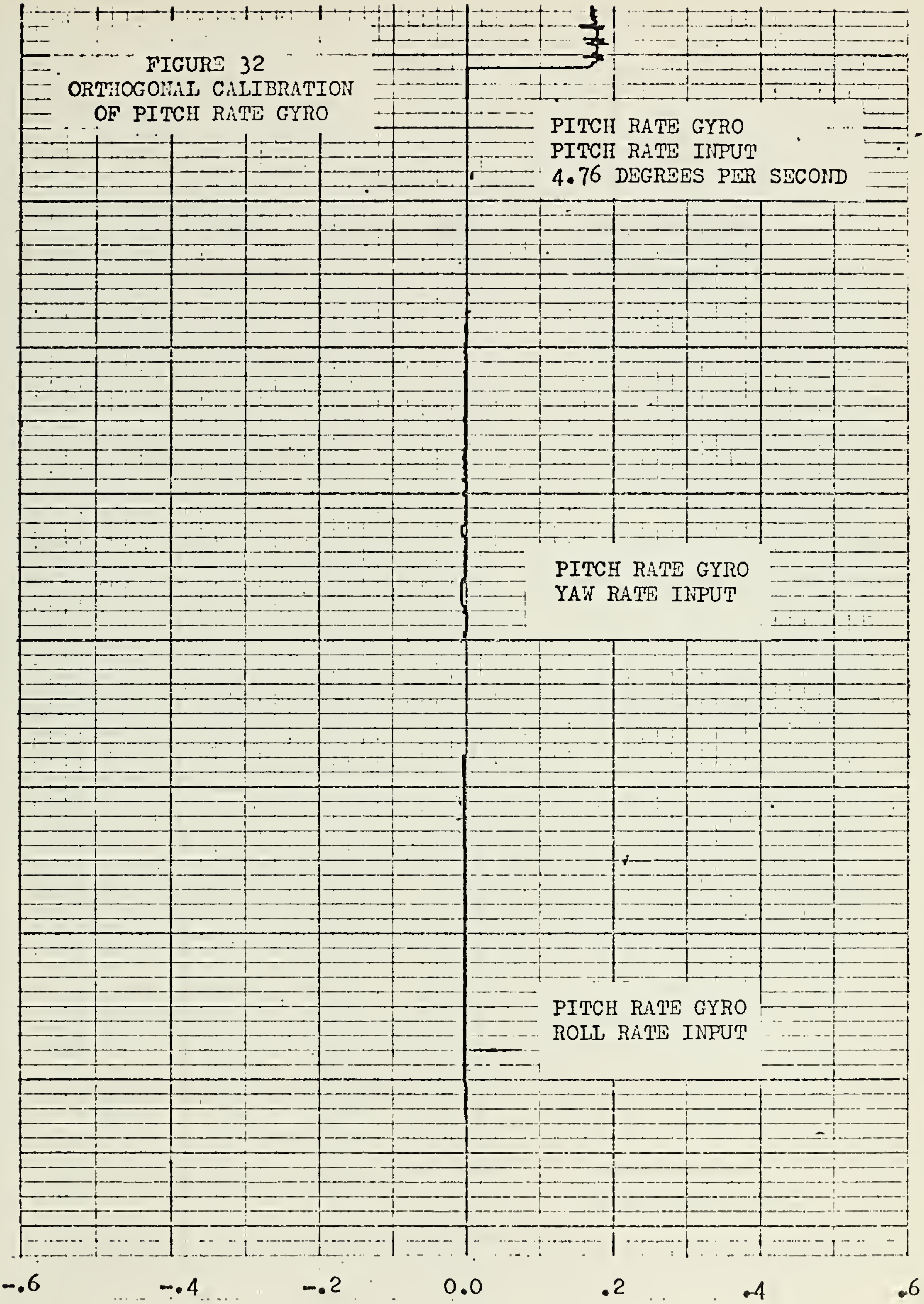


FIGURE 32
ORTHOGONAL CALIBRATION
OF PITCH RATE GYRO

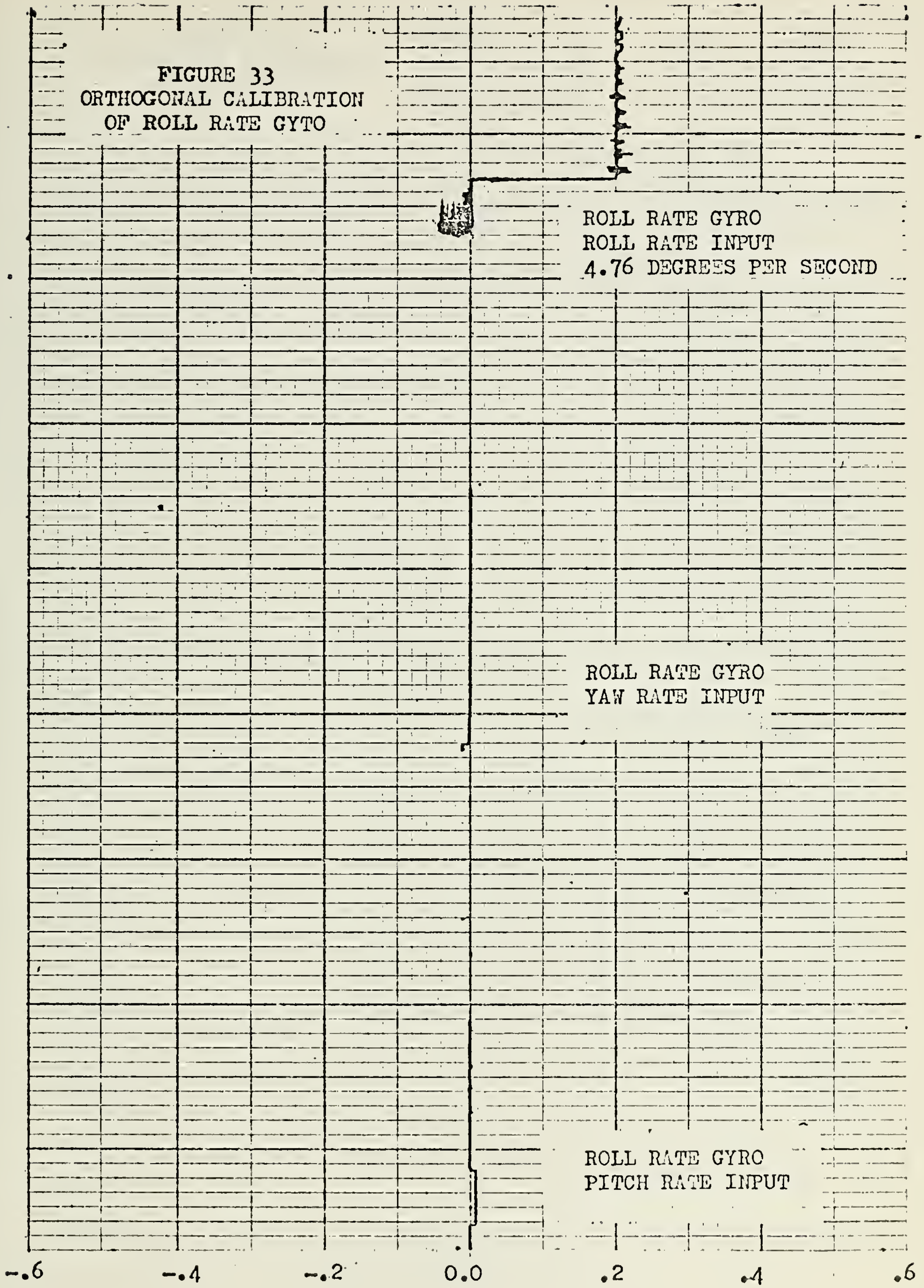


FIGURE 33
ORTHOGONAL CALIBRATION
OF ROLL RATE GYTO
ROLL RATE GYRO
ROLL RATE INPUT
4.76 DEGREES PER SECOND
ROLL RATE GYRO
YAW RATE INPUT
ROLL RATE GYRO
PITCH RATE INPUT
-.6
-.4
-.2
0.0
.2
.4
.6

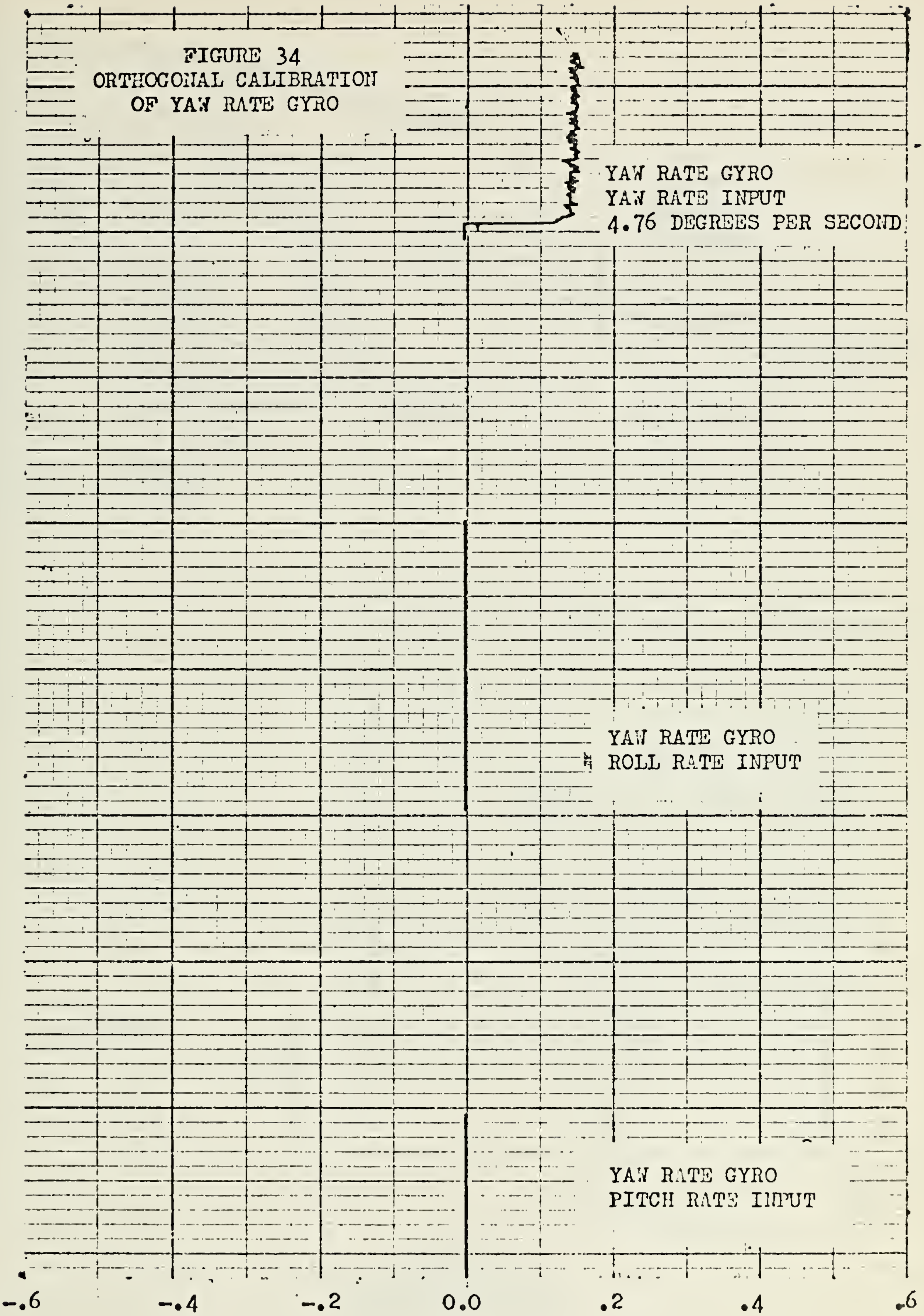


FIGURE 34
ORTHOGONAL CALIBRATION
OF YAW RATE GYRO
YAW RATE GYRO
YAW RATE INPUT
4.76 DEGREES PER SECOND
YAW RATE GYRO
ROLL RATE INPUT
YAW RATE GYRO
PITCH RATE INPUT
-.6
-.4
-.2
0.0
.2
.4
.6

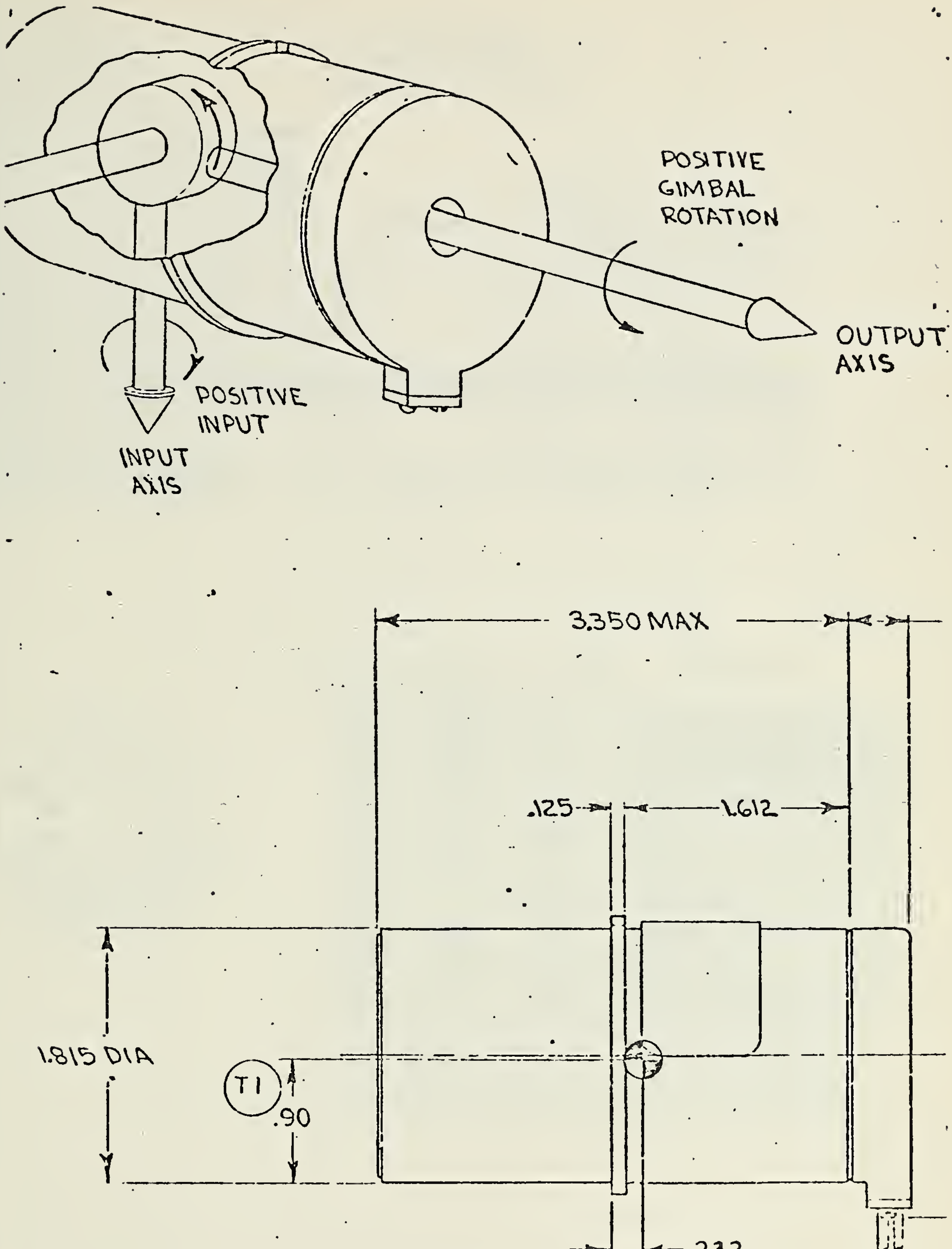
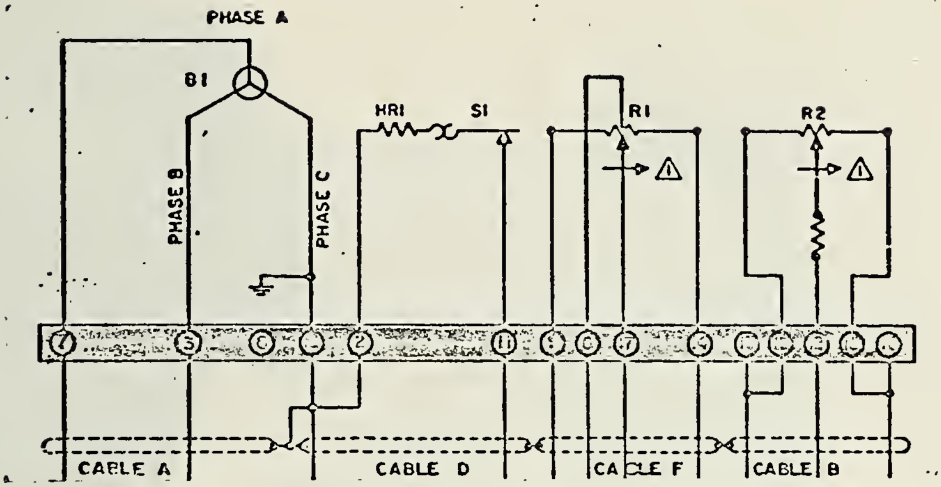
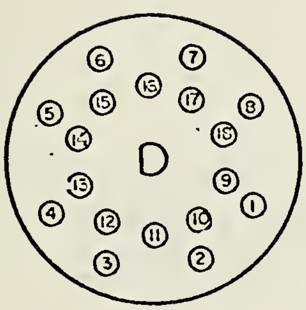


FIGURE 35
DIMENSION FOR A
GG16C14 RATE GYRO

FIGURE 36
SCHEMATIC DIAGRAM FOR
A GG16C14 RATE GYRO





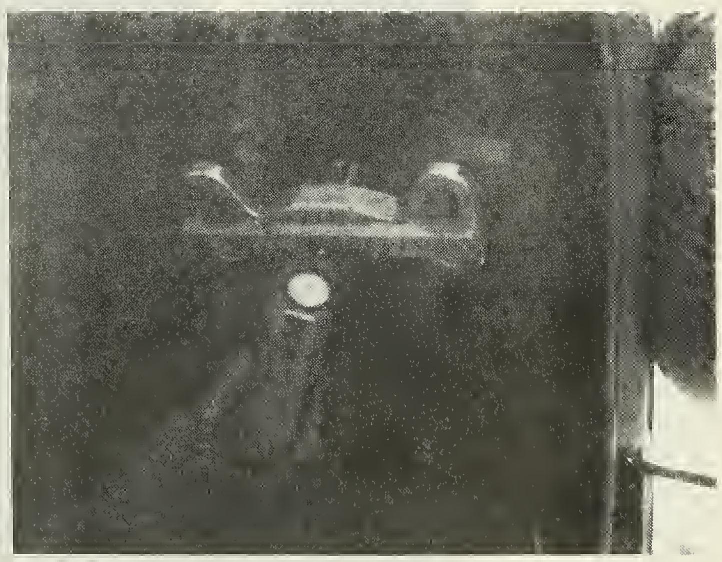
COLOR CODE PRIMARY	STRIPE	TO TERM NO.	FUNCTION
WHT	YEL	7	SPINMOTOR-PHASE A
WHT	VIO	3	SPINMOTOR-PHASE B
WHT	BLK	6	COM SPINMOTOR LEAD
WHT	GRN	12	WIPER-POT. 2
WHT		13	POT. 2
WHT	BLU	18	POT. 2
RED		2	HTR
BLU		16	HTR GRD
WHT	BRN	17	WIPER POT. 1
WHT	ORN	9	POT. 1
WHT	RED	14	POT. 1
GRN		8	POT. 1 CENTER TAP

FIGURE 37

AILERON FORCE
CALIBRATION
TECHNIQUE

FIGURE 38

ELEVATOR FORCE
CALIBRATION
TECHNIQUE



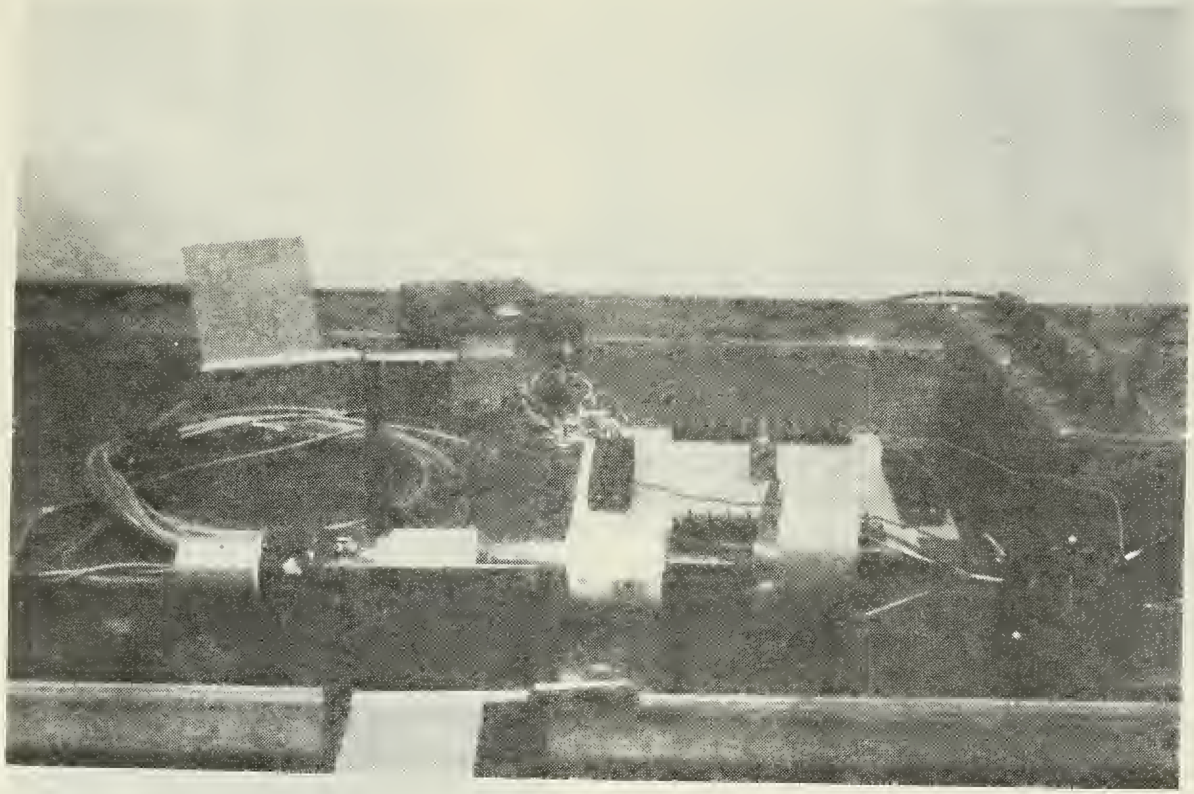


FIGURE 39 YAPS VANE CALIBRATION SET

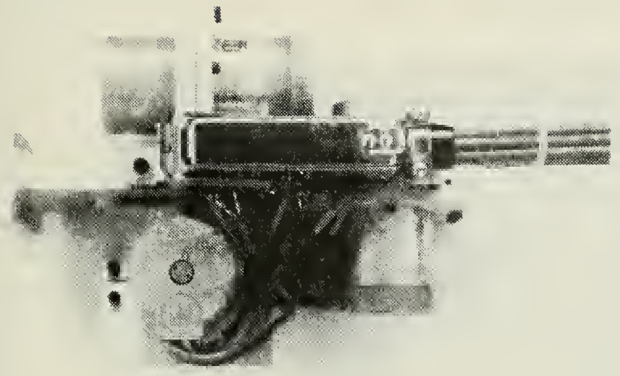


FIGURE 40

RATE GYRO
PACKAGE

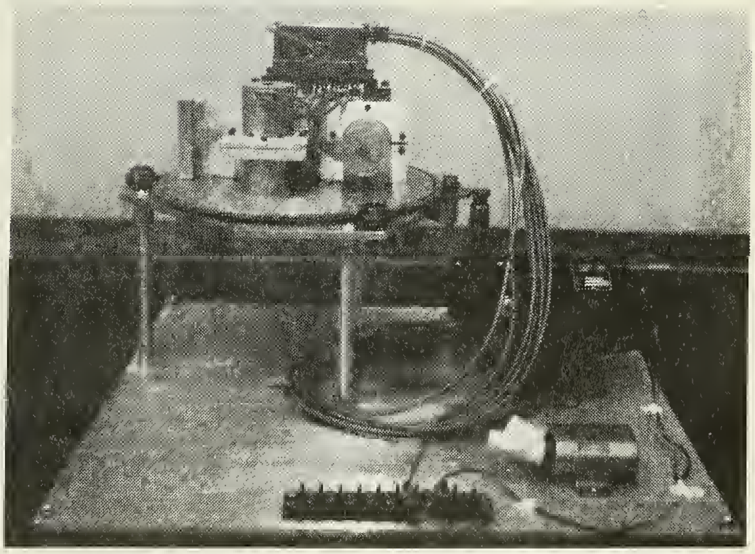


FIGURE 41

RATE GYRO
PACKAGE ON
TURN TABLE

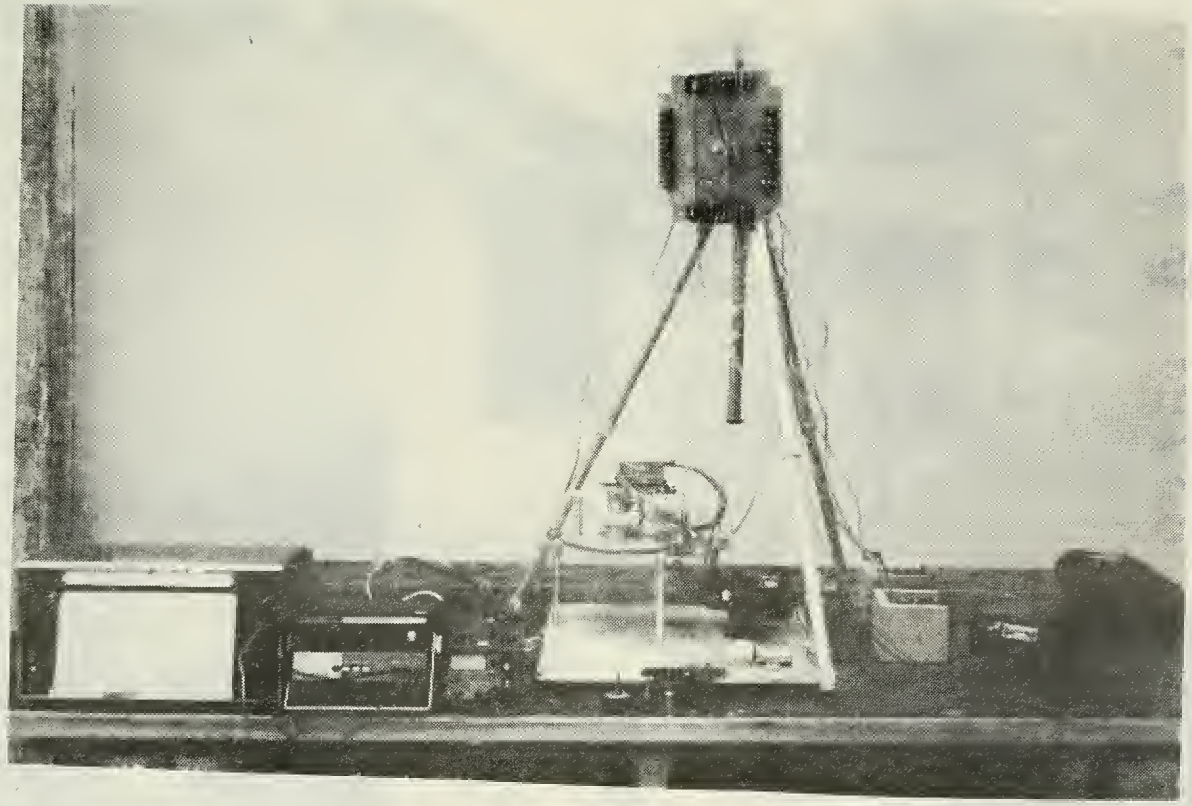


FIGURE 42 RATE GYRO PACKAGE CALIBRATION SET-UP

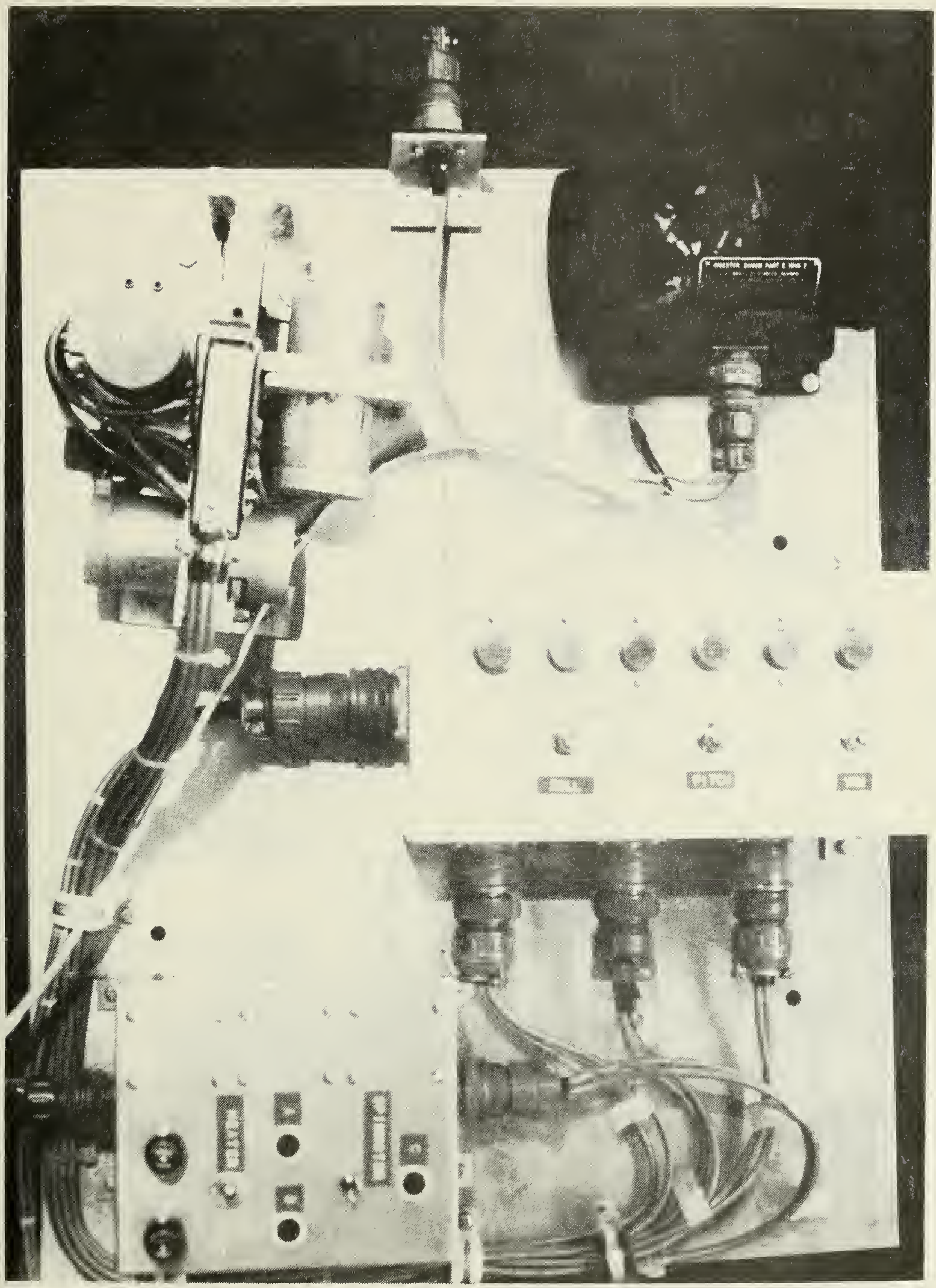
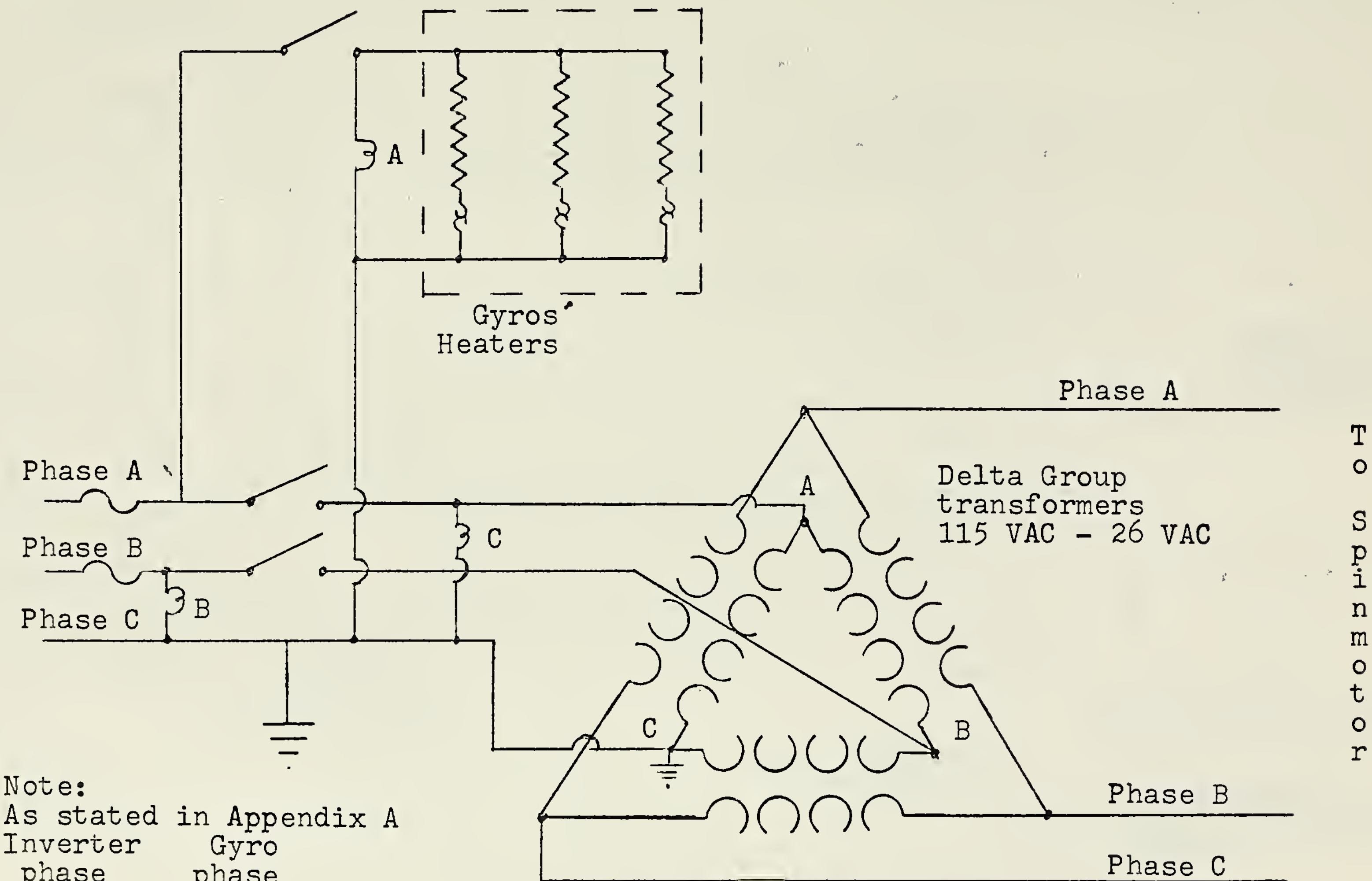


FIGURE 43 AIRCRAFT RATE GYRO PACKAGE UNIT



Note:
As stated in Appendix A

Inverter phase		Gyro phase
A	=	A
C	=	B
B	=	C

See Appendix A for wire color code

FIGURE 44 AC DISTRIBUTION CHASSIS

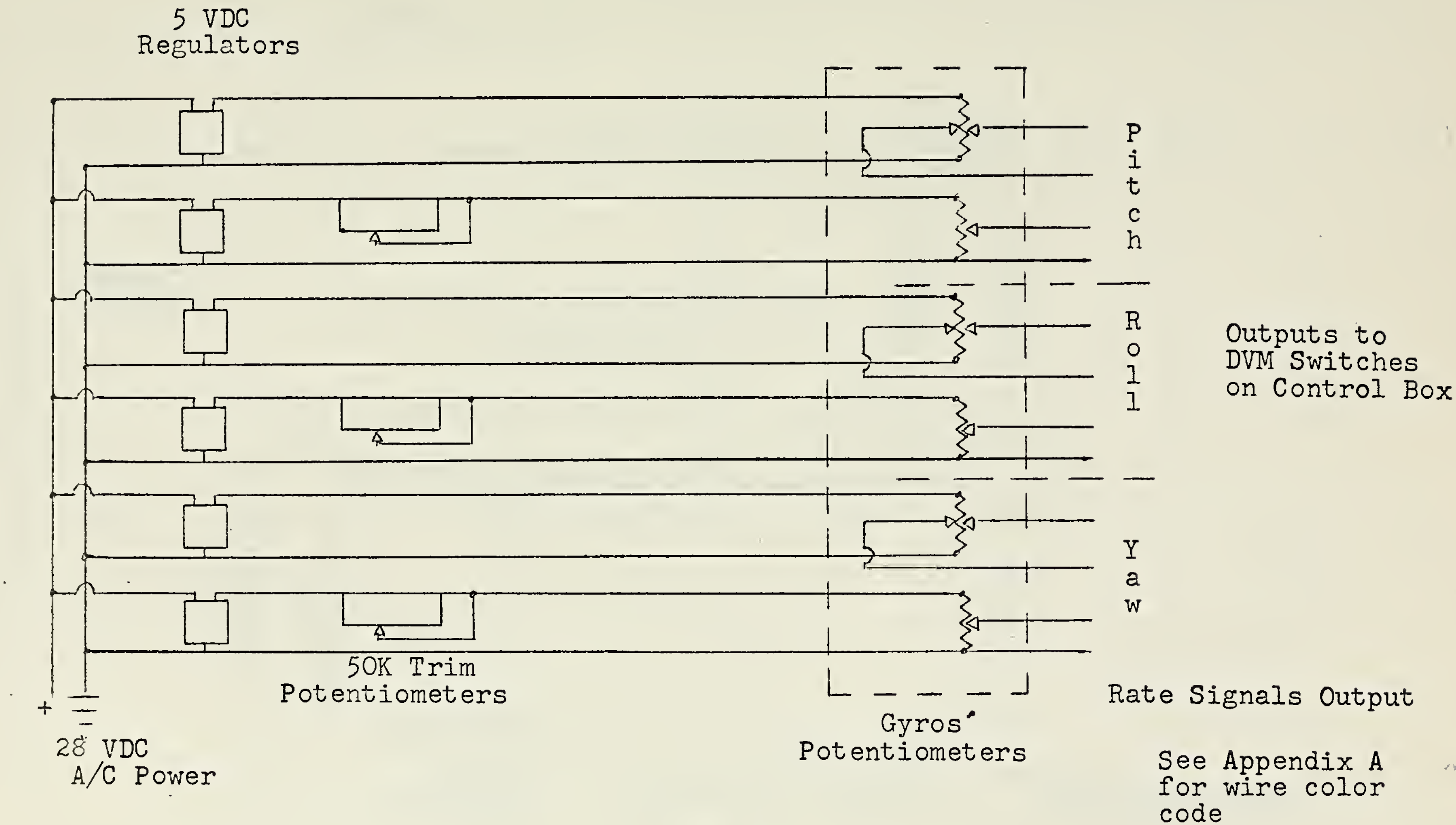


FIGURE 45 DC DISTRIBUTION CHASSIS

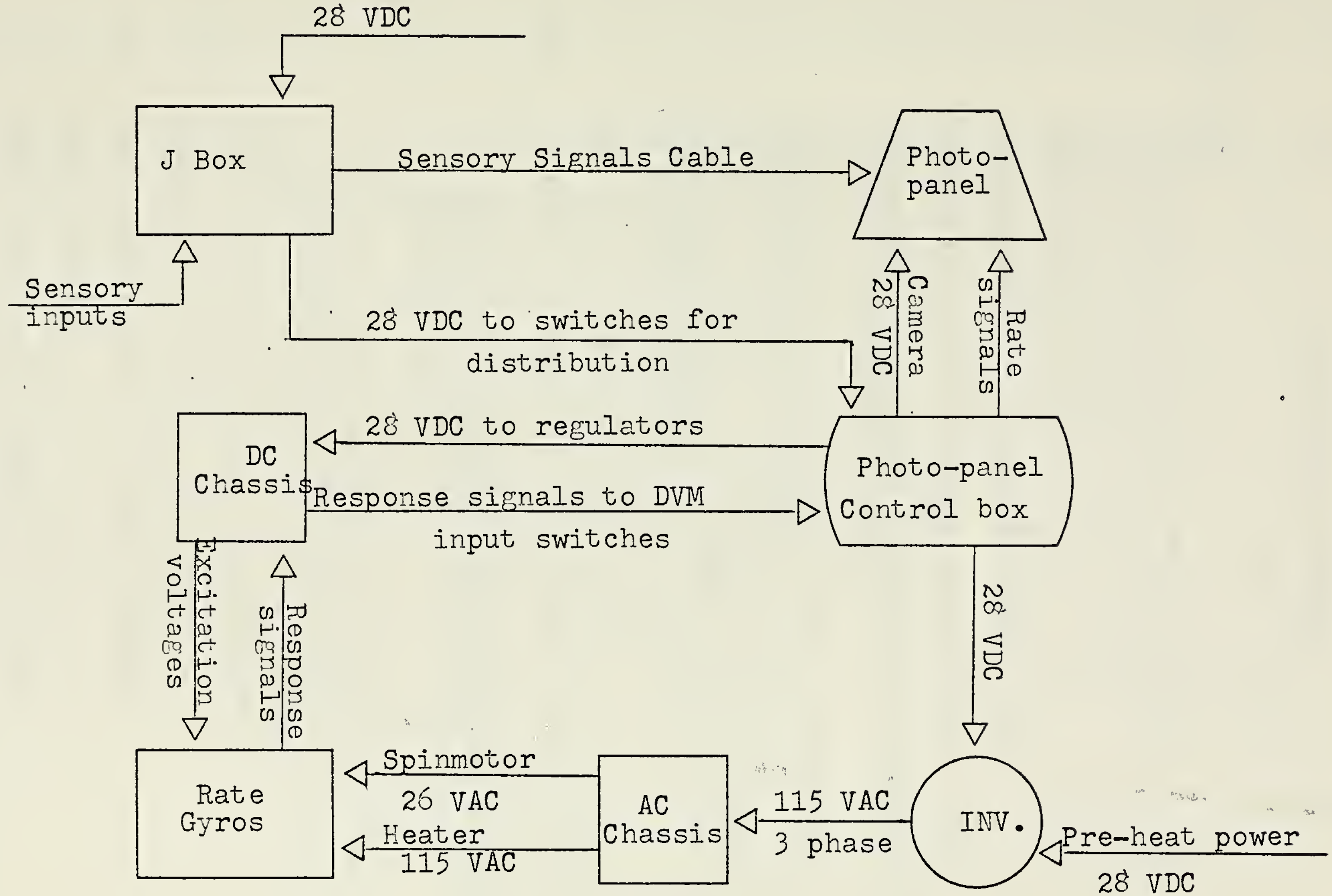


FIGURE 46 DATA ACQUISITION AND RATE GYRO PACKAGE SYSTEMS

APPENDIX C

Operating Procedures

Preflight:

1. Check photo-panel power switch - OFF.
2. Check inverter power switch - OFF.
3. Check regulator power switch - OFF
4. Check AC chassis power switches - OFF.
5. Check digital panel-meter power switch - OFF.
6. Check that wire bundle is installed between aircraft junction box and photo-panel.
7. Check security of pressure tubing connections on underside of photo-panel.
8. Connect regulated power supply set at 28 VDC to the input power receptacle of the junction box. Adjust to 5.0 AMP output.
9. Junction box power switch - ON.
10. Perform zero position balancing on all meters:
    a. Control deflection meters, angle of attack and side slip instruments are zeroed by adjusting individual balancing potentiometer nuts on junction box.
    b. Control force meters are zeroed by adjusting the operational amplifier zero setting as indicated.
11. Open hinged access door on photo-panel.
12. Set event counter in photo-panel.
13. Set event counter on control panel.
14. Set altimeter.
15. Wind and actuate elapsed time clock.
16. Photo-panel power switch - ON.

17. Check photo-panel lights for illumination.

18. Gust lock - INSTALLED.

19. Apply aileron, elevator and rudder forces to respective co-pilot controls to check movement of control force gauges in photo-panel.

20. Gust lock - REMOVED.

21. Actuate all controls and check movement of control deflection gauges in photo-panel.

22. Secure hinged access door.

23. Photo-panel power switch - OFF.

24. Junction box power switch - OFF.

25. Disconnect external power source and reconnect aircraft plug to junction box.

26. Install loaded camera on photo-panel:

    a. Ensure camera is centered on mount.

    b. Route shutter release cable to clam shell fitting at solenoid and secure.

    c. Close camera mount and tighten wing nut sufficiently to secure camera. (Do not overtighten)

27. Check camera lens settings:

    a. Range - 2 feet.

    b. F-stop - 4.0.

    c. Shutter speed - 125.

28. Connect 28 VDC external power supply to rate gyro package pre-heat plug. (Check DC ammeter in external cable circuitry.)

29. Pre-heat procedures:

    a. 28 VDC external power switch - ON. (Adjust to 10 AMPS.)

    b. AC chassis heater power switch - ON.

    c. At this point light "A" and "B" should be ON; if not, secure the rate gyro package by positioning heater power switch to - OFF.

30. Monitor DC ammeter: When amperage drops from 8 to 10 down to 1 AMP, pre-heating is complete. Allow 15 to 20 minutes for this evolution.

31. AC chassis heater power switch - OFF.

32. Secure and disconnect external power. (Heater power must be re-energized within approximately 10 minutes.)

After Engines Started:

33. Junction box power switch - ON.

34. Inverter power switch - ON.

35. Regulator power switch - ON.

36. AC chassis heater power switch - ON.

37. AC chassis spinmotor power switch - ON.

38. Ensure that all three lights on AC chassis are illuminated, if not, secure the rate gyro system.

Inflight Turn On:

39. Photo-panel power switch - ON.

40. Activate shutter button once regardless of ready light indication. (This properly sequences camera with control system.)

41. Actuate film advance button.

42. Digital panel-meter power switch - ON.

43. Select appropriate digital panel-meter inputs.

44. Record data as desired. (Observe ready light cautions)

Inflight Secure:

45. All control panel power switches - OFF. *

46. All rate gyro package power switches - OFF. *

47. Junction box power switch - OFF. *

* Must be accomplished prior to engine secure.

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